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Design Considerations for Composite Fuselage Structure of Commercial Transport Aircraft

G.W. Davis and I.F. Sakata

LOCKHEED-CALIFORNIA COMPANY
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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
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FOREWORD

This report documents the results of a study performed by the Lockheed-California Company under subcontract to the Lockheed-Georgia Company for the National Aeronautics and Space Administration (NASA), Langley Research Center (LaRC), Hampton, Virginia. The effort was directed to the identification of design considerations that could impact the design of a composite material fuselage structure and to delineate the principal design drivers. The study was conducted for the NASA LaRC Structural Mechanics Branch under Contract NAS1-15949, Task Assignment No. 1.

J. N. Dickson of the Lockheed-Georgia Company was the Program Manager of the Advanced Composite Structure Design Technology program. I. F. Sakata was Lockheed-California Company Project Leader and G. W. Davis, Principal Investigator. Dr. J. H. Starnes, Jr., was the NASA Technical Monitor. The following Lockheed-California Company employees also made significant contributions to the material contained in this report:

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J. Soovere	Sonic Fatigue
R. H. Stone	Composite Repair
J. Van Hamersveld	Producibility
B. Saelman	Weights

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DESIGN CONSIDERATIONS FOR COMPOSITE FUSELAGE STRUCTURE OF COMMERCIAL TRANSPORT AIRCRAFT

By G. W. Davis and I. F. Sakata

Lockheed-California Company

SUMMARY

A study was conducted to explore the structural, manufacturing, and service and environmental considerations that could impact the design of composite fuselage structure; to assess the severity of these considerations; and to delineate the principal design drivers. A summary of the major design considerations discussed in this report are listed in Table 1. Each consideration is ranked with respect to whether it is a principal design driver, a requirement that probably will not govern the design but should be checked (secondary requirement), or a consideration that requires the development of new design criteria.

INTRODUCTION

The design of a fuselage for a commercial transport is impacted by the interaction of its functional requirements and its basic strength, stiffness, and life requirements. Functional systems, such as the ingress and egress systems, passenger accommodations (seats, windows, lavatories, etc.), environmental control, and cargo containment interact with and modify the basic design features of the fuselage structure. In addition, provisions must be made in the design for the interface requirements of the nose landing gear, the wing/fuselage interface structure, the flight station and the empennage.

These multifaceted requirements impose severe restrictions on the basic configuration of the shell and the structural-material concepts selected for use in its design. New and innovative designs must be explored to accommodate these requirements and to meet the goals of lower weight and more cost-effective structure for future airplanes. Considerable weight saving potential is forecast with the application of composite materials to the fuselage of commercial transports. However, before this can become a reality, a state of design readiness must be attained that includes (1) a thorough understanding of the problems associated with the design of a composite fuselage, (2) a delineation of the major design problems, and (3) the development of the necessary design data base to assess and solve these problems. This report addresses the first phase of design readiness, identifies the major design considerations and discusses their impact on the design of composite fuselage structure.

TABLE 1. - SUMMARY OF PRINCIPAL DESIGN DRIVERS FOR COMPOSITE FUSELAGES

Considerations	Principal Design Drivers	Secondary Requirements	Requires New Criteria or Methodology	Comments
Structural Considerations				
Airplane Weight	√			Large weight savings are being forecast. Comprehensive design studies on large composite components are required to validate weight equations.
Shell Size		√		Generally an economic consideration, could impact buckling and stiffness.
Fuselage Stiffness		√		Could affect the static aeroelastic behavior and the flutter speed.
Temperature/Humidity		√	√	Interpretation of data required, degradation of material strength must be accounted for in allowables.
Lightning		√	√	New design practices are required.
Hail	√		√	New criterion and an assessment of the impact damage required.
Shell Cutouts	√			Doors, windows and cutouts will affect the general arrangement of the shell.
Joints	√			Joint design requires a detailed knowledge of the local stresses and the load distribution of the fasteners.
Frame/Stringer Intersection		√		Because of the relatively low interlaminar tension and shear values of current graphite/epoxy materials, mechanical fasteners or stitching will most likely be required on pressurized fuselages.
System Interface Requirements		√		Detail design studies required to assess the design problems associated with these considerations.
Structural Interface Requirements		√		
Minimum Skin Thickness		√		Minimum skin thickness based on manufacturing, and damage tolerance and fail-safe considerations.
Loads		√		The aeroelastic behavior and the requirements for emergency landing could be influenced by the added stiffness of composite structure.
Material Properties		√	√	An improved resin system would improve durability aspects of current materials.
Design Strain Levels		√		Criterion required to quantify the effects of cutouts, joints, impact damage and transverse cracking on the design strain level.
Buckling Limitations	√			Design development required to establish post-buckling limits for weight efficient shell design.
Damage Tolerance Requirements	√		√	Realistic impact criteria must be formulated to establish fatigue and fail-safe policies.
Acoustic Transmission	√			With a reduction in shell mass the design of the structure and the interior noise control elements must be explored to control the noise transmissibility.

TABLE 1. - SUMMARY OF PRINCIPAL DESIGN DRIVERS FOR COMPOSITE FUSELAGES (Continued)

Considerations	Principal Design Drivers	Secondary Requirements	Requires New Criteria or Methodology	Comments
Structural Considerations (Continued)				
Sonic Fatigue		√		Fatigue life depends on the design details of the composite structure. Testing program required to establish S/N curves for various types of structure.
Crashworthiness	√			Development of data base is essential in order to design a composite fuselage from inception to meet crashworthiness goals.
Manufacturing Considerations				
Materials/Material Cost	√		√	An advanced resin system with improved physical and processing requirements could greatly impact the material and fabrication costs. Extensive use of woven cloth and preplied material forms to reduce costs.
Fabrication Costs	√		√	Minimum bleed control systems, simplified cure cycles, automated roll-forming, cutting and layup machines, and the use of more cocured assemblies.
Tooling Requirements		√		Development of tooling methods to produce cocured skin/stiffener assemblies.
Equipment Requirements		√	√	Develop automatic production machines to minimize the handwork labor.
Fabrication Procedure		√	√	Develop automatic production machines and control equipment.
Producibility	√		√	Design laminates and shapes amenable to automatic production, maximum use of cocuring, reduce fastener count, and utilize preplied tape materials.
Service and Environmental Considerations				
Safety and Reliability	√			Airframe design criteria must be established to ensure airplane life and meet all requirements of FAA, the manufacturer, and the airlines.
Maintainability	√			Airline damage results primarily from impact, fatigue and corrosion with the lower fuselage the most damage-prone area. Most impact damage is from ground handling. Composites are expected to eliminate corrosion and reduce fatigue damages.
Inspectability	√		√	The composite fuselage structure must be designed for visual inspections by airline personnel. NDI procedure required to verify the extent of damage need development.
Repairability	√		√	Repair procedures must be developed for composite fuselage designs. These procedures must be compatible with airlines capabilities and restore design strength and fatigue life of the structure.

STRUCTURAL CONSIDERATIONS

The design of a composite fuselage must provide the necessary strength and rigidity to sustain the loads and environment that it will be subjected during the operational life of the aircraft. The many structural considerations must adhere to the requirements defined in the Federal Aviation Regulation, Part 25 (Reference 1) in order to achieve the objectives of 1) unlimited life in operational service and 2) fail-safe characteristics for any reasonable extent of damage. The advisory circulars also sets forth guidance information relating to acceptable means of compliance with the provisions of FAR 25 dealing with composite structures (Reference 2) and with damage tolerance and fatigue evaluation certification requirements (Reference 3).

These many requirements impose severe constraints on the design of the fuselage structure. The major structural considerations are presented to indicate the general policy and type of data required to establish criteria for composite fuselage structure design.

General Requirements

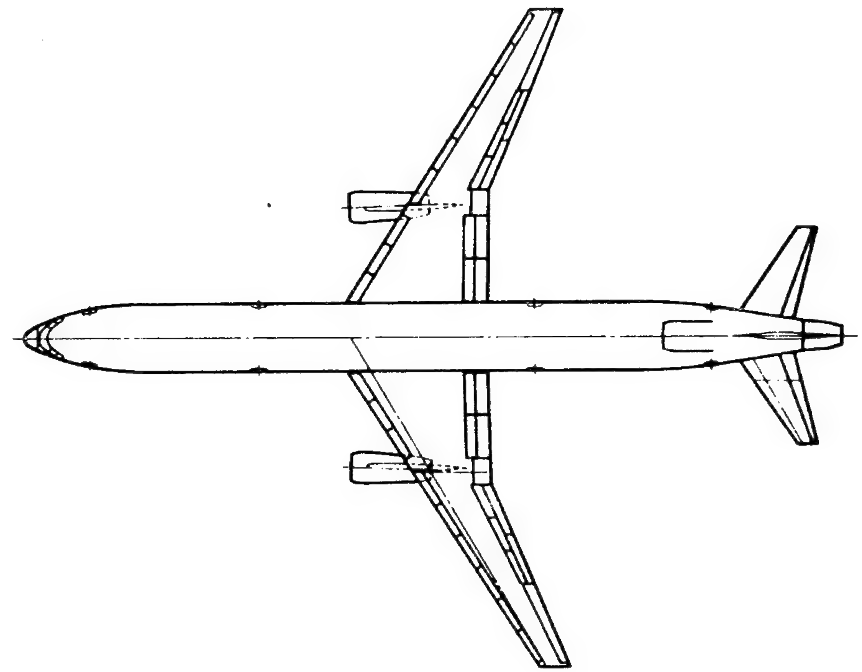
The general arrangement of an advanced technology transport aircraft is shown in Figure 1. This transport incorporates three advanced, mixed-flow, turbofan engines, a supercritical wing with reduced leading-edge sweep, the use of composite material for both primary and secondary structure, and active controls. As noted on this figure, this airplane has a wing semispan of 27.74 m (94.3 ft) and a fuselage length of 70.0 m (229.7 ft). In addition, this airplane has a 331 m^2 (3558 ft^2) wing planform area with a gross weight at takeoff of 183,970 kg (405,500 lbm). This configuration has a payload of 36,290 kg (80,000 lbm), equivalent to 400 passengers, and a range of 5560 km (3000 n.mi). Table 2 summarizes the airplane characteristics.

The weights assigned to the various components of the baseline airplane are listed in Table 3. The two largest structural weight items are the wing and body. These items amount to 19,650 kg (43,118 lbm) and 24,940 kg (54,991 lbm), respectively. The fuselage represents approximately 14 percent of the airplane weight at takeoff. A more detailed weight statement of the composite fuselage design is presented in Table 4 and indicates that 20,784 kg (45,820 lbm) is attributed to primary structure, which is 83 percent of the total fuselage weight. The corresponding fuselage weight of an equivalent advanced technology aircraft that uses aluminum for its basic material is also shown in this table. The composite fuselage design indicates a weight saving of approximately 21 percent over the more conventional fuselage design.

Fuselage shell sizes are dictated by aircraft size and passenger seating arrangement, performance, and structural optimization. The fuselage diameters of existing and new commercial aircraft are shown in Figure 2. For future aircraft, only slight variations in fuselage diameter are expected.

CHARACTERISTICS	WING	HORIZONTAL TAIL	VERTICAL TAIL
AREA SQ. M. (SQ. FT.)	330.54 (3558)	71.54 (770.1)	44.51 (479.1)
ASPECT RATIO	10	5	1.6
SPAN M. (FT.)	57.49 (188.6)	18.93 (62.1)	8.12 (27.68)
ROOT CHORD M. (IN.)	8.85 (348.25)	5.82 (228.96)	8.12 (319.54)
TIP CHORD M. (IN.)	2.65(104.47	1.74 (68.68)	2.43 (95.86)
TAPER RATIO	0.30	0.30	0.30
MAC M. (IN.)	6.31(248.28)	4.15 (163.7)	5.72 (225.18)
SWEEP RAD., (DEG.)	0.524 (30)	0.524 (30)	0.611 (35)
T/C ROOT %	12	10	10
T/C TIP %	12	10	10

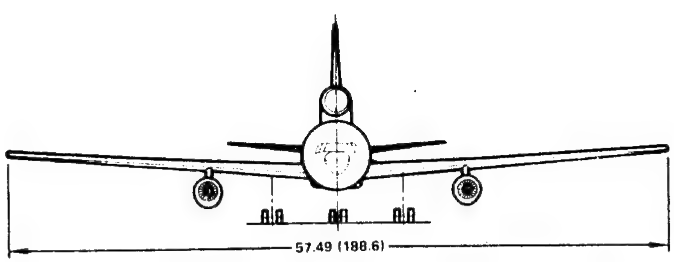
GROSS WEIGHT · 183977 KG (405592 LB M.)
POWER PLANT · ADVANCED MIXED FLOW TURBOFAN
INSTALLED THRUST · 154550 N (34746 LB. F.)
PASSENGERS · 400
RANGE · 3,000 N.M.
GEN. ARR. · 3,000 N.M.
400 PAX. ADV. TURBOFAN



2. DIMENSIONS IN METERS (FEET), OR NOTED

1. CADAM REF. DWG. CL1337-1-1, 1, 2, 3

NOTE



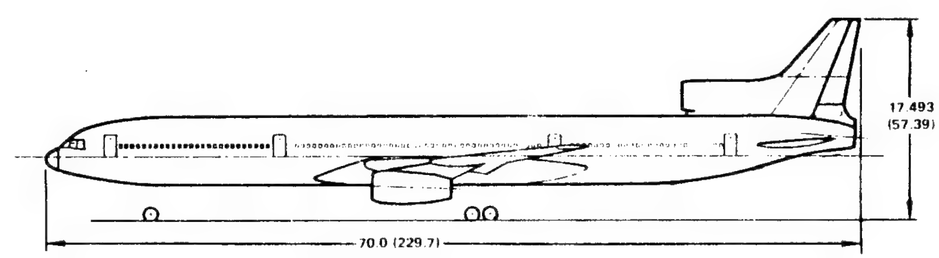


Figure 1. - General arrangement, RE-1011 advanced technology airplane.

TABLE 2. - AIRPLANE CHARACTERISTICS

Aircraft Model	RE-1011
Wing Area – m^2 (ft^2) Overall Length – m (ft) Wing Span – m (ft) Overall Height – m (ft)	330.5 (3 558) 70.0 (229.7) 57.5 (188.6) 17.5 (57.4)
Operational Weights – kg (lbm) Maximum Takeoff Maximum Zero Fuel Operating Empty	 183 970 (405 590) 142 940 (315 130) 106 650 (235 130)
Payload – kg (lbm)	36 290 (80 000)
Engine Model Takeoff Thrust – N (lbf)	Advanced Mixed-Flow Turbofan 154 560 (34 750)
Range – km (n.mi.)	5 560 (3 000)

TABLE 3. - RE-1011 AIRPLANE GROUP WEIGHT STATEMENT

Item	Mass	
	(kg)	(lbm)
Wing	19 558	43 118
Tail	2 211	4 875
Body	24 943	54 991
Landing Gear	7 800	17 196
Surface Controls	1 951	4 301
Nacelle and Engine Section	2 644	5 830
Propulsion	13 254	29 219
Auxiliary Power Unit	506	1 116
Instruments	393	867
Hydraulics	1 099	2 423
Electrical	2 651	5 844
Avionics	998	2 200
Furnishing and Equipment	16 671	36 754
Environmental Control System	3 484	7 682
Deicing System	181	398
Mfg. Empty Weight (MEW)	98 345	216 814
Std. and Oper. Equip.	8 307	18 314
Operating Empty Weight (OEW)	106 652	235 128
Payload	36 287	80 000
Zero Fuel Weight (ZFW)	142 939	315 128
Fuel	41 034	90 464
Takeoff Weight	183 973	405 592

TABLE 4. - RE-1011 FUSELAGE WEIGHT BREAKDOWN

Item	RE-1011	
	Composite Design kg (lbm)	Aluminum Design kg (lbm)
Skin (inc. doublers, joints)	7 873 (17 357)	10 225 (22 542)
Stringers and Longerons	1 712 (3 774)	2 224 (4 902)
Frames	1 976 (4 357)	2 567 (5 659)
Floor Supports (inc. seat track)	3 527 (7 775)	4 580 (10 098)
Flooring	1 231 (2 713)	1 464 (3 228)
Keelson Web	1 200 (2 646)	1 558 (3 436)
Pressure Decks	436 (961)	566 (1 248)
Bulkheads	2 829 (6 237)	3 674 (8 100)
Fuselage Primary Structure	20 784 (45 820)	26 858 (59 213)
Secondary Structure (Windshield, windows fairing, radome, etc.)	4 160 (9 171)	4 842 (10 675)
Total Fuselage	24 943 (54 991)	31 701 (69 888)

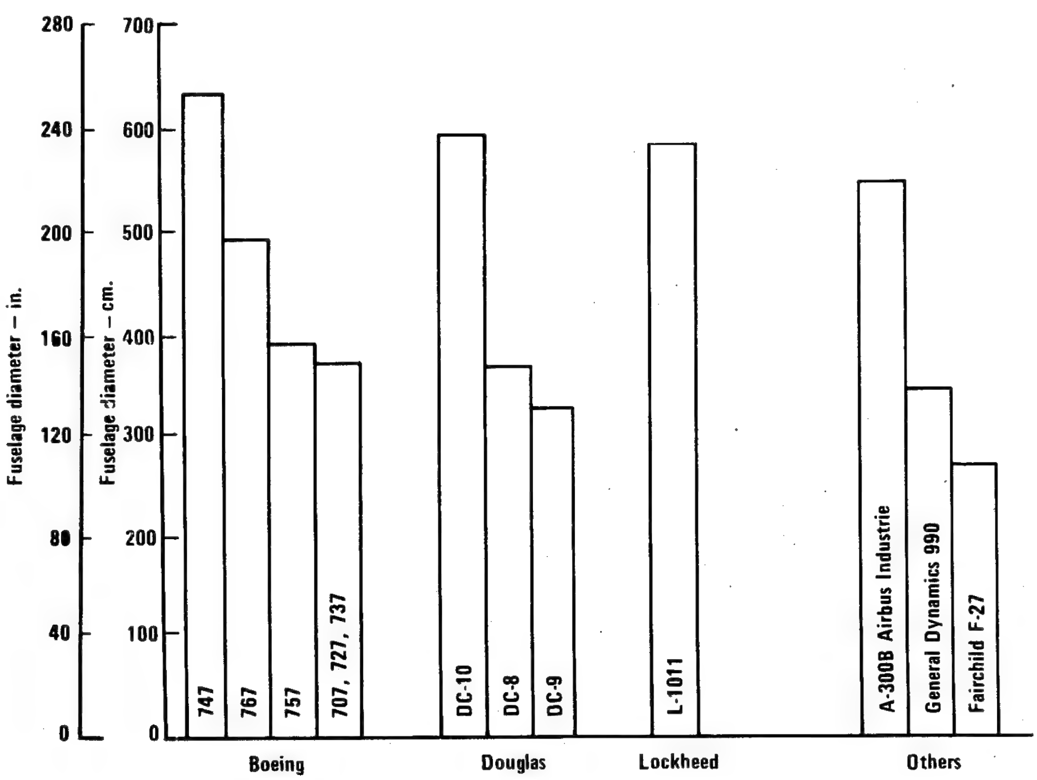


Figure 2. - Fuselage diameters for commercial transport aircraft.

Changes in fuselage diameter can affect the buckling and postbuckling behavior of the shell; the method of fabrication and the ease in handling of the shell components and assemblies during fabrication; the magnitude of the membrane forces due to cabin pressurization and the corresponding minimum skin gage; and the overall bending, shear and torsional stiffness of the shell.

Changes in the stiffnesses of the fuselage shell can impact both the static aeroelastic behavior and the elastic dynamic modes of the airplane. The effectiveness of the control surfaces on the tail can be affected by the elastic deformation of the fuselage afterbody. In addition, the stiffness of afterbody could affect the frequency of the vibration modes and the critical flutter speed. The overall bending and torsional stiffnesses for a typical aluminum fuselage of a wide-bodied airplane are presented in Figure 3.

Environmental Requirements

The sensitivity of composite materials to environmental conditions imposes problems that are generally either not considered or not encountered in the design of conventional metal aircraft. Some of the more important environmental considerations on composite structure are: temperature/humidity, lightning and hail. These environmental conditions are discussed in the following text.

Temperature/Humidity.- Temperature and humidity histories to which an aircraft will be exposed must be considered in depth. Climatological data have been collected from many areas of the world and should be used to help in the establishment of the design criteria. For example, temperature exceedance data of selected U.S. cities are presented in Figure 4. The interpretation of the data, however, presents some problems. These problems include the reasonableness of using extremes in temperature and humidity data or average data. Temperature and humidity profiles for individual airplanes may vary considerably, depending on the route structure. Accordingly, some airplanes may be exposed to severe temperature and humidity conditions more often than other airplanes in the fleet. This difference in exposure must be accounted for in a rational manner in the establishment of design criteria.

The climatological data, once established, must be used in conjunction with the composite material emissivity and absorption qualities to establish the temperature and humidity levels which must be used in determining the composite material strength levels and allowables to be used for design.

Other factors that must be considered include the effects of prolonged exposure to direct sunlight and high humidity while the aircraft is sitting on the ground in still air. Certain areas of the structure will attain higher temperatures than others, such as the upper surface of the fuselage

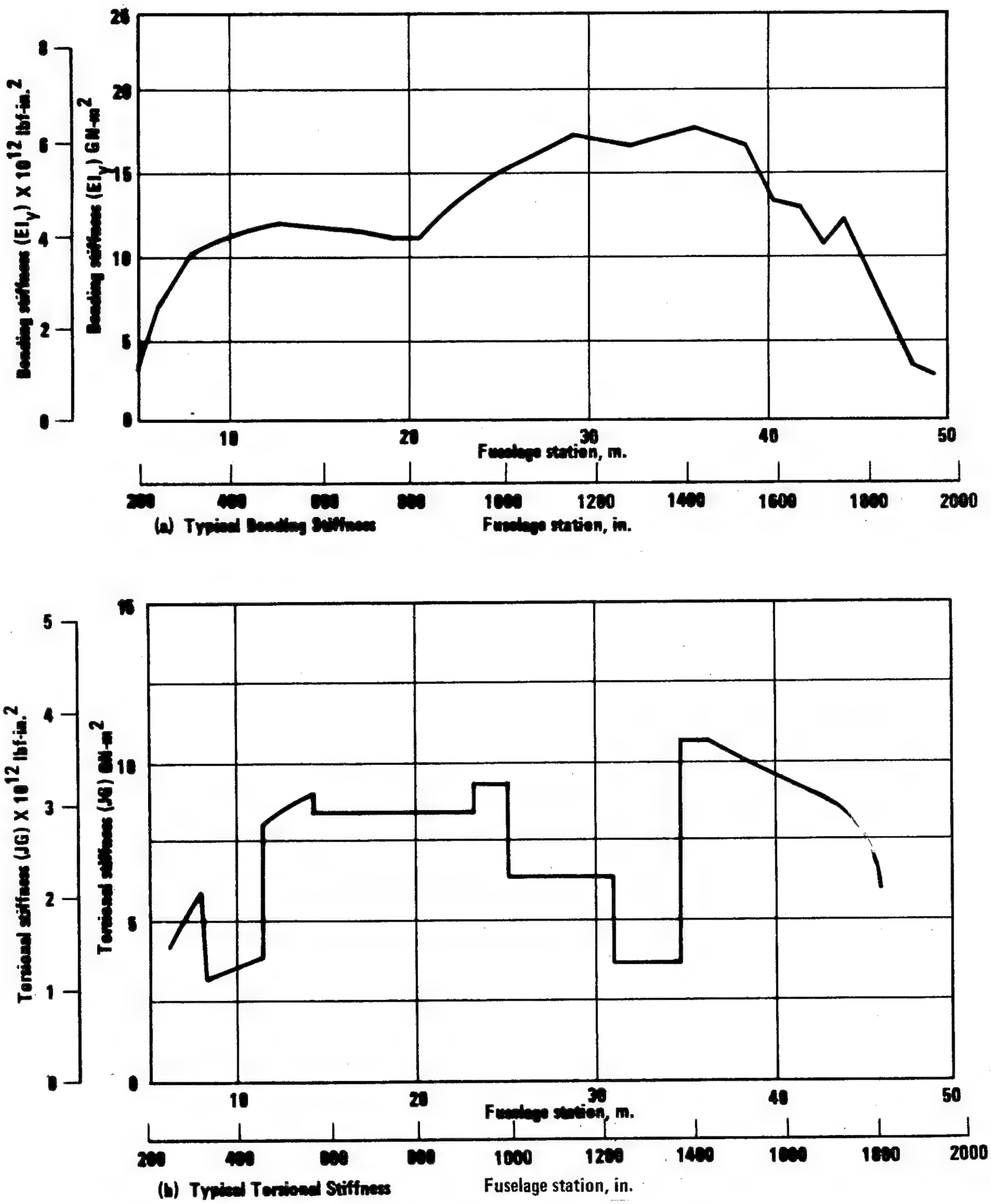


Figure 3. - Typical fuselage stiffnesses for a wide-bodied aluminum aircraft.

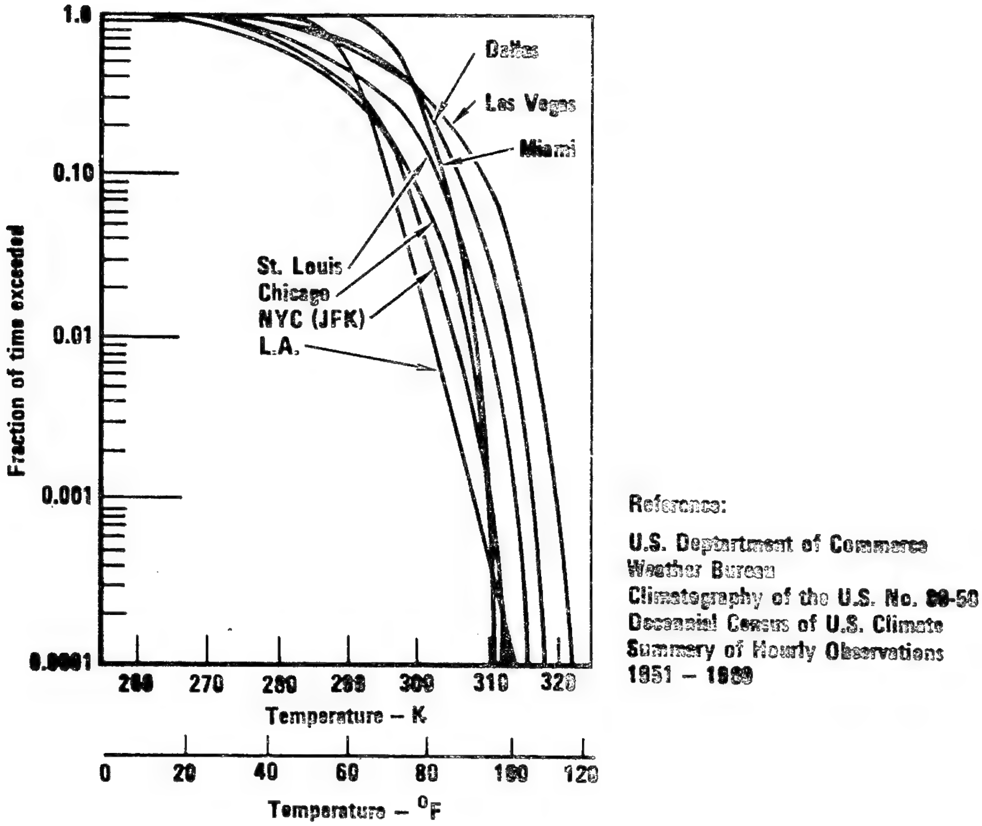


Figure 4. - Fraction of time exceeded temperature of selected U.S. cities.

versus the lower surface. The presence of reflective surfaces or other external heat sources in the proximity of the airplane must also be considered.

An analysis was conducted to assess the effects of solar heating on the structural temperature of a representative composite fuselage structure. The fuselage geometry corresponded to that of the L-1011 airplane with a typical material distribution being defined for a skin/stringer design using the T300/5208 graphite/epoxy material system. Fuselage surface temperatures were calculated at the upper and lower crown locations on the forebody. These surfaces were analyzed for two surface coatings: a sprayed aluminum coating and a dark colored paint. Solar absorptivity and emissivity values of 0.50 and 0.20 were used for the sprayed aluminum coating and respective values of 0.80 and 0.90 for the painted surface.

The maximum skin temperatures attained on the upper and lower surfaces after an hour exposure to sunlight on the ground at an ambient temperature of 318 K (112°F) are shown in Figure 5. These temperatures are attained when the surfaces are painted black or dark blue. The upper crown structure with this coating achieves a steady-state maximum skin temperature of 379 K (223°F) with a corresponding temperature on the lower crown structure of

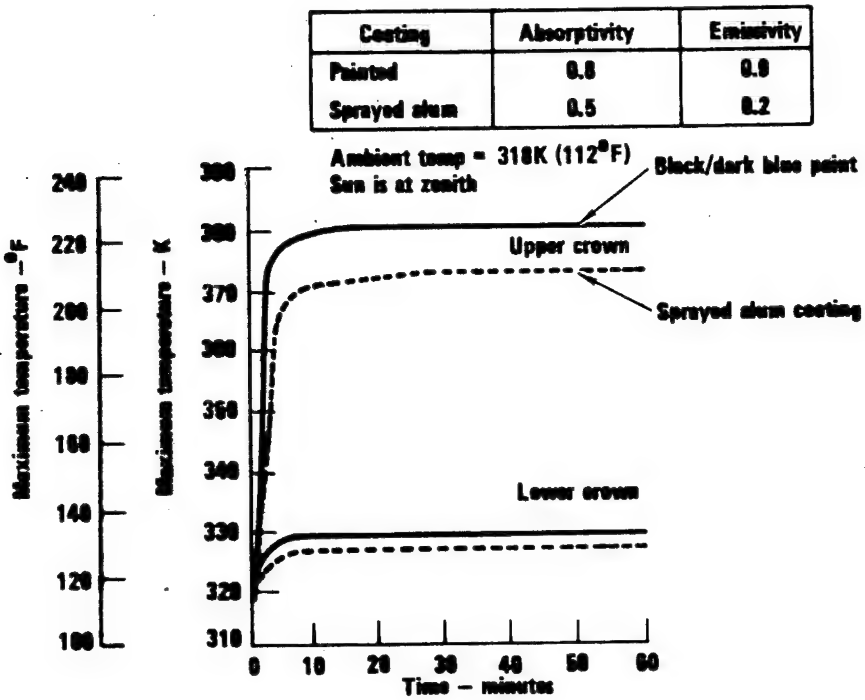


Figure 5. - Ground transient solar heating for upper crown of composite fuselage.

329 K (132°F). For the sprayed aluminum coating the maximum temperatues were 7 K (13°F) cooler on the upper crown and 2 K (4°F) cooler on the lower crown.

Solar heating analyses have been performed on various composite structural components postulated for application to the airframe of the L-1011 aircraft. These components include the inboard aileron, the vertical fin, the wing box and the fuselage shell. Results of these temperature analyses are summarized in Table 5 and include a brief description of the structure and its surface preparation.

Lightning.- The application of composite structures reduces the inherent electromagnetic shielding and lightning-current-carrying capabilities achieved with electrically continuous aluminum designs. Most composite structures have some electrical conductivity but can be damaged structurally by high current flow through the fibers. The protection design concept must prevent lightning current from attaching to or tranferring through the composite structures.

TABLE 5. – MAXIMUM SURFACE TEMPERATURES DUE TO SOLAR HEATING(1)

COMPONENT	STRUCTURAL DESCRIPTION	SURFACE CONDITION	SUN ATTITUDE	AMBIENT TEMPERATURE	MAXIMUM SURFACE TEMPERATURES		COMMENT
					SUN-FACING SURFACE	SHADOW EXPOSED SURFACE	
Composite Inboard Aileron	Honeycomb sandwich surface panels with ribs and spars. All elements use T300/5208 Gr/E material.	Aluminum spraycoat. Solar absorptivity = 0.5, Emissivity = 0.2	Zenith	318K (112^{0}F)	371K (209^{0}F)	–	Maximum temperature occurs at mid-panel on the outer face sheet of the upper surface.
Composite Vertical Fin Structure	Two spar structural arrangement with hat-stiffened surface panels. Surface panels and spar caps fabricated using T300/5208 Gr/E material.	Painted black or dark blue Solar absorptivity = 0.8 Emissivity = 0.9	45^{0}	318K (112^{0}F)	362K (192^{0}F)	–	Direct solar heating of the fin and heating due to reflection from the horizontal stabilizer are accounted for in the analysis.
Composite Wing Box (Outboard wing Station)	Multi-rib structural arrangement with integrally stiffened wing surface panels. T300/5208 Gr/E material.	Aluminum spray coat Solar absorptivity = 0.5 Emissivity = 0.2	Zenith	318K (112^{0}F)	369K (205^{0}F) (no fuel) 352K (175^{0}F) (full fuel)	347K (165^{0}F) (no fuel) 312K (102^{0}F) (full fuel)	Maximum surface temperatures reflect wing box geometry and panel cross section data at OWS 452. Fuel temperature = 311K (100^{0}F).
Composite Fuselage (Forebody)	Skin/Stringer shell with frames. I-stiffener laminates T300/5208 Gr/E material.	Aluminum spray coat or dark paint. See above absorptivity and emissivity values	Zenith	318K (112^{0}F)	379K (223^{0}F) (paint) 372K (210^{0}F) (aluminum spray)	329K (132^{0}F) (paint) 326K (128^{0}F) (aluminum spray)	Maximum surface temperatures occur when the shell is painted. An insulation thickness of 5.72 cm (2.25 in.) was used for the analysis. No cooling from the ECS system was considered.

1. After one hour of exposure to sunlight on the ground in still air at the specified ambient temperature.

Some promising developmental lightning protection methods that should be considered are aluminum diverter strips, aluminum wire mesh, and aluminum flame spray. Knowledge gained through the Advanced Composite Vertical Fin (ACVF) and Advanced Composite Aileron (ACA) programs, other ACEE composite structures programs, and industry, NASA, Air Force and Navy research programs should be used in designing the overall lightning protection configuration.

Since the entire aircraft becomes a radiating antenna at some frequencies, special consideration also must be given to electrical bonding and noise interference from precipitation static charging during the design of the lightning protection system.

Existing electromagnetic design practices, when applied to composite structures are, for the most part, unworkable due to the low conductivity and lack of shielding effectiveness of composite materials and joints. Composite materials exhibit considerable reduced shielding properties when compared to aluminum (Reference 4). Figures 6 and 7 show some typical measured values of both magnetic and electric field shielding available from graphite and boron composite structures relative to that provided by aluminum. These curves should not be considered absolute but merely as trends, since the available shielding depends on many factors, e.g., material dimensions and grounding. This implies that the susceptibility due to lightning effects will be many times more severe. It is important not to place the burden of providing equivalent performance to aluminum on composite structures, because the benefits gained will be seriously compromised. Replacing metal structures with composite structures will require that new concepts for integrating avionic systems be evaluated.

The development of the lightning protection system for the avionics and fuel systems will be two of the more important elements of the entire protection program, not only because of safety but also because of the difficulty in arriving at designs which will meet the present FAA and CAA lightning protection requirements.

Composite structure must be tested to verify the lightning protection design and to evaluate electromagnetic field penetration at the joints and also through the composite material. Some antenna and fuel system component installations must also be tested.

<u>Hail.</u>- A likely source of objects that can cause damage to the fuselage is hail. Figure 8 presents the terminal velocity of free-falling hail at sea level conditions (Reference 5). Damage from this source could occur on the ground on the upper surface or in flight on the upper, side and lower surfaces of the fuselage.

In addition to the size, terminal velocity, and probability, the number of hailstones impinging on a composite fuselage of an airplane per unit area

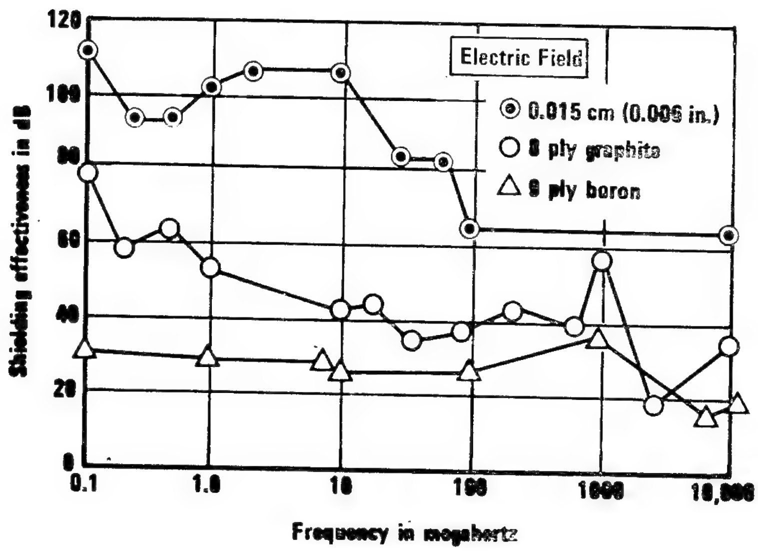


Figure 6. - Shielding effectiveness in an electric field.

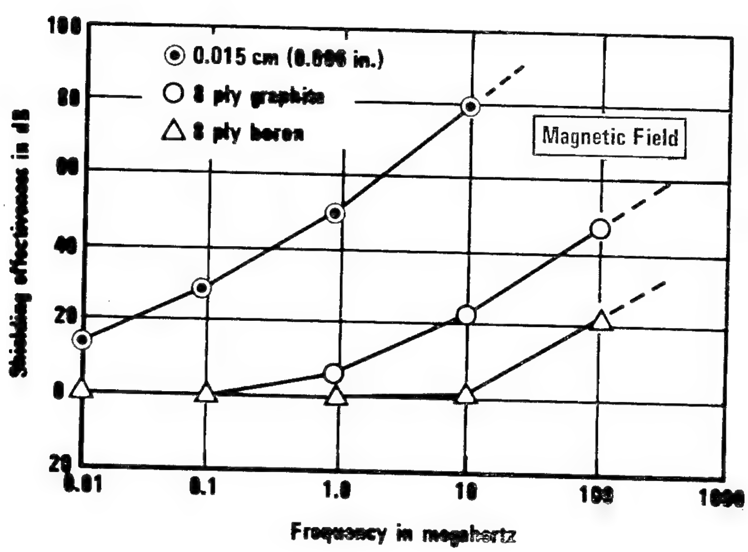


Figure 7. - Shielding effectiveness in a magnetic field.

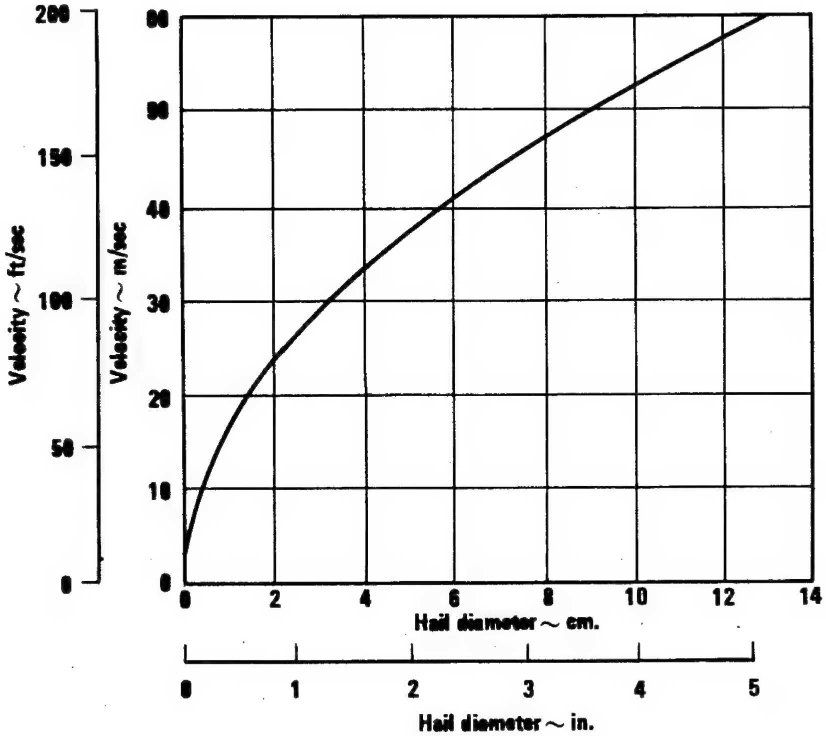


Figure 8. - Hail terminal velocity, sea level to 1.52 km (5000 ft).

as a function of duration may also be of importance for both ground and flight operations. For instance, a single impact from a large-size hail may produce nondetectable localized damage for which, on a one-time basis, the reduced strength could be tolerated until the next inspection period. However, the impingement of small-size hail on the damaged area may cause further strength loss which cannot be tolerated. A preliminary assessment of the type of data required to define a hail criterion is presented in the damage-tolerance section.

Basic Design Requirements

The design of a fuselage will be impacted by its functional as well as its basic strength, stiffness and life requirements. Functional systems such as the ingress and egress systems, passenger accommodations (seats, windows, lavatories, etc.), environmental control system, and cargo containment interact with the basic design features of the fuselage structure. In addition, the shell must be designed to accommodate the interface requirements of the nose landing gear, the wing carry-through structure, and the flight station and empennage.

The impact of some of these considerations on the structural requirements of a wide-bodied aircraft fuselage design and their possible influence on the design of a composite fuselage are discussed below.

The fuselage of the L-1011 airplane (Figure 9), is a conventional semi-monocoque structure fabricated using aluminum alloy materials, and has a circular cross section 5.97 m (235 in.) in diameter for the major portion of its length. This constant section, the flight station and a small section where the fuselage begins to taper at the aft end form the fuselage pressure shell. This pressure shell is designed for the pressure differential attained with an 2.44 km (8000 ft.) altitude cabin pressure at an airplane altitude of 12.8 km (42000 ft.).

Shell Cutouts and Holes.- The composite fuselage design must allow for the same types of penetration of the basic shell as a conventional aluminum design (Figure 10). The L-1011 fuselage contains eight plug-type passenger doors of which six are main entry doors, 1.07 m (42 in.) wide by 1.93 m (76 in.) high, and the other doors measure 0.61 by 1.52 m (24 x 60 in.). In addition, doors are required for access to the various cargo compartments; on the L-1011 airplane a maximum opening of approximately 1.78 by 1.73 m (70 by 68 in.) is provided for the forward and center cargo compartments.

Cabin windows, located midway between the fuselage frames, are provided at approximately 0.508 m (20 in.) spacing throughout most of the cabin length of the L-1011 fuselage. These windows are located on the sidewall of the fuselage and are mounted in window frame forgings that are riveted to the skin and a bonded doubler. Figure 11 shows a sidewall panel with several window installations.

The location and size of these doors and windows will affect the geometry and spacing of the frame and stringer design of a composite shell as it does a conventional aluminum design. Reinforcement members must be provided around these cutouts to avoid any needless discontinuities in the structure and to allow for an efficient transfer of load. Metallic reinforcement members may be required in areas where high concentrated loads occur. In these areas, strain compatibility with the adjacent composite structure must be maintained to ensure that the fatigue quality and accompanying life requirements are met.

Joints.- The size of current wide-bodied aircraft requires a large number of longitudinal and girth joints for subassembly and assembly of the fuselage and its structural components. A typical fuselage barrel section splice joint for the L-1011 is shown in Figure 12. This shear-type joint incorporates both a short stringer doubler (approximately one-bay long) and a skin doubler. Skin splice joints, in addition to maintaining the pressure integrity of the shell, must sustain the flight and landing loads imposed on

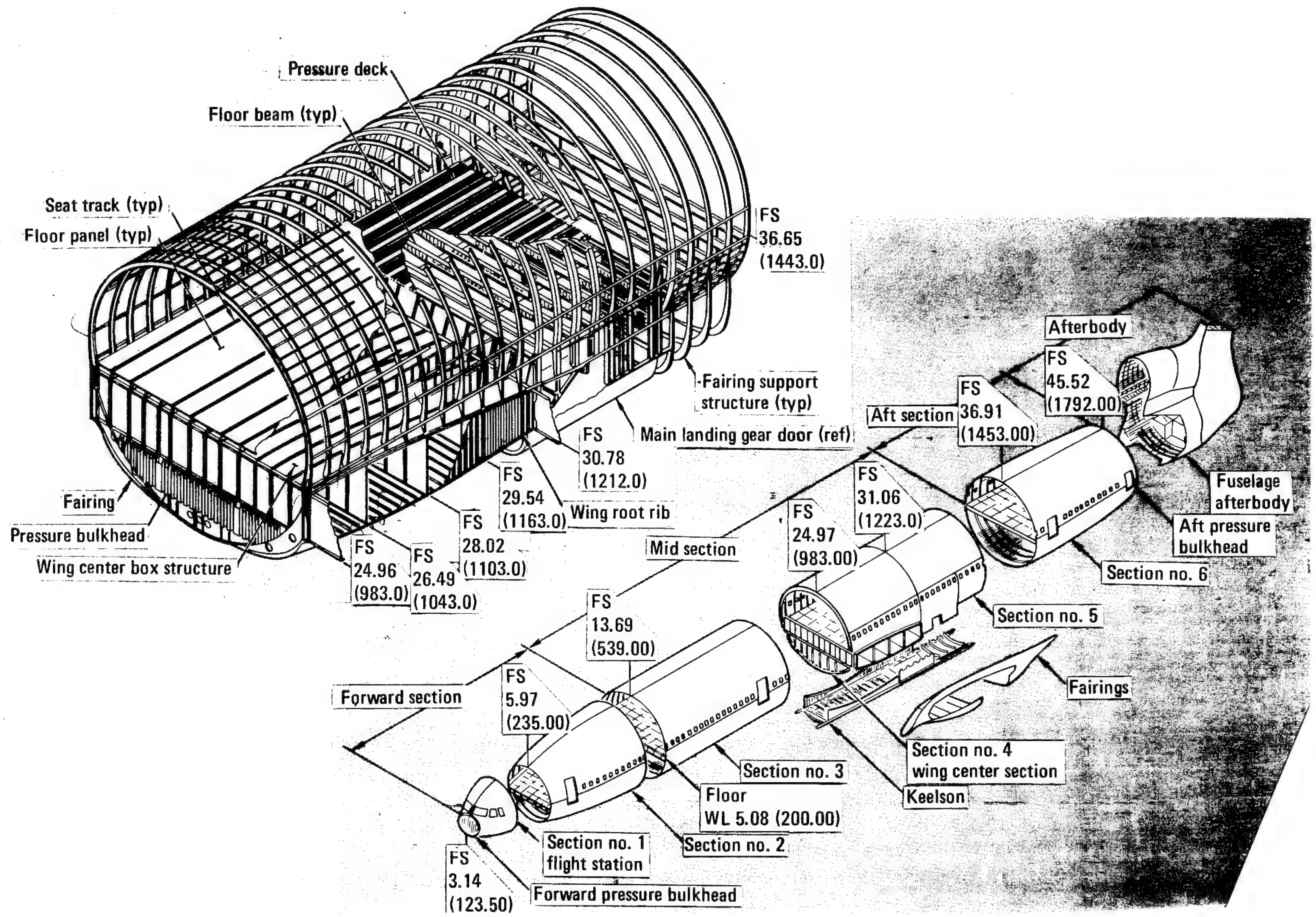


Figure 9. - Fuselage structural arrangement.

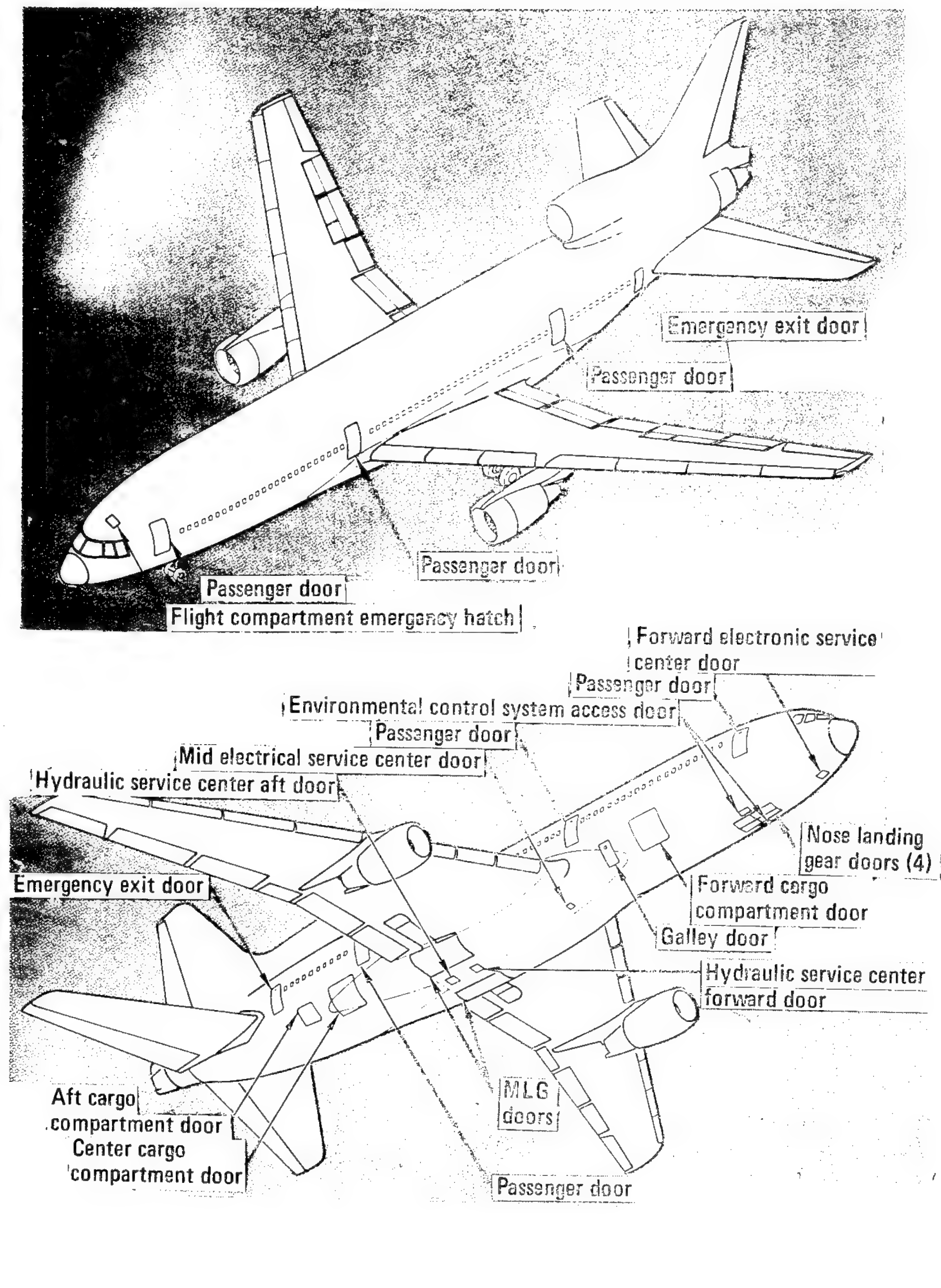


Figure 10. - Door locations.

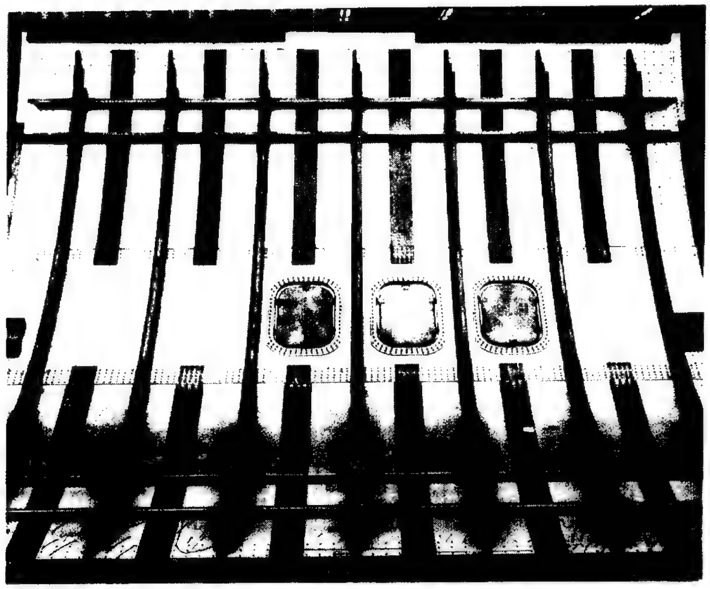


Figure 11. - Typical window installation.

Figure 12. - Typical fuselage barrel splice joint.

the aircraft during its operational lifetime. In general, the critical design loading for a longitudinal joint is the hoop forces due to pressurization, whereas, the combined forces due to pressurization and body bending are more critical for a girth joint.

In addition to the efficient transfer of these loads, another basic objective in the design of joints is that its fatigue life should be extemely long and that if cracking occurs it initiates outside the joint area, preferably in the basic panel. This requires a detailed knowledge of the local stresses in the joint area. This stress state is very much dependent on load distribution by the fastener system, bearing load distribution through the fastener hole, eccentricity of the splice members and flexing of the support structure.

To achieve the required structural integrity in a composite joint design, care should be exercised so that: (1) only close tolerance fasteners are used, (2) fasteners are selected for their corrosion resistance, (3) no unsupported splice joints are permitted, and (4) the effects of deflections, moisture induced expansion, and thermal expansion of adjacent connected structure are considered in the design. In addition to these considerations, only laminate layups that minimize interlaminar shear and tension stresses at the edges should be used.

Frame/Stringer Intersection.- The design of a fuselage structure must provide the necessary strength and rigidity to sustain the appropriate pressurization loads in combination with the basic body-bending loads imposed during the operational life of the aircraft. Current metallic fuselage designs and the majority of the proposed composite designs incorporate shells of skin/stringer configuration with internal frames for reinforcing the shells. This type of construction, although beneficial in many aspects, creates discontinuity forces at the juncture of the shell and frame when the cabin is pressurized. These forces cause the shell to pillow out between frames, because the radial growth of the shell under pressurization is being restrained by the adjacent frames. This pillowing effect, combined with the requirements for general instability, for distribution of concentrated frame loads, and for damage tolerance design, dictates the method of attachment and design of the frame/shell interface. Typical designs of this area for several commercial aircraft are shown in Figure 13.

For composite fuselage structure, the interface forces at the shell/frame juncture impose more stringent requirements on the design than that of a comparable aluminum structure because of the relatively low interlaminar tension and shear properties of laminated graphite/epoxy structure. Skin shear ties and stringer clips are most likely required in areas of high compression and/or shear loading on the pressurized cabin. The use of mechanical fasteners and/or other attachment methods, such as stitching, is advisable for the design of the frame/stringer intersection.

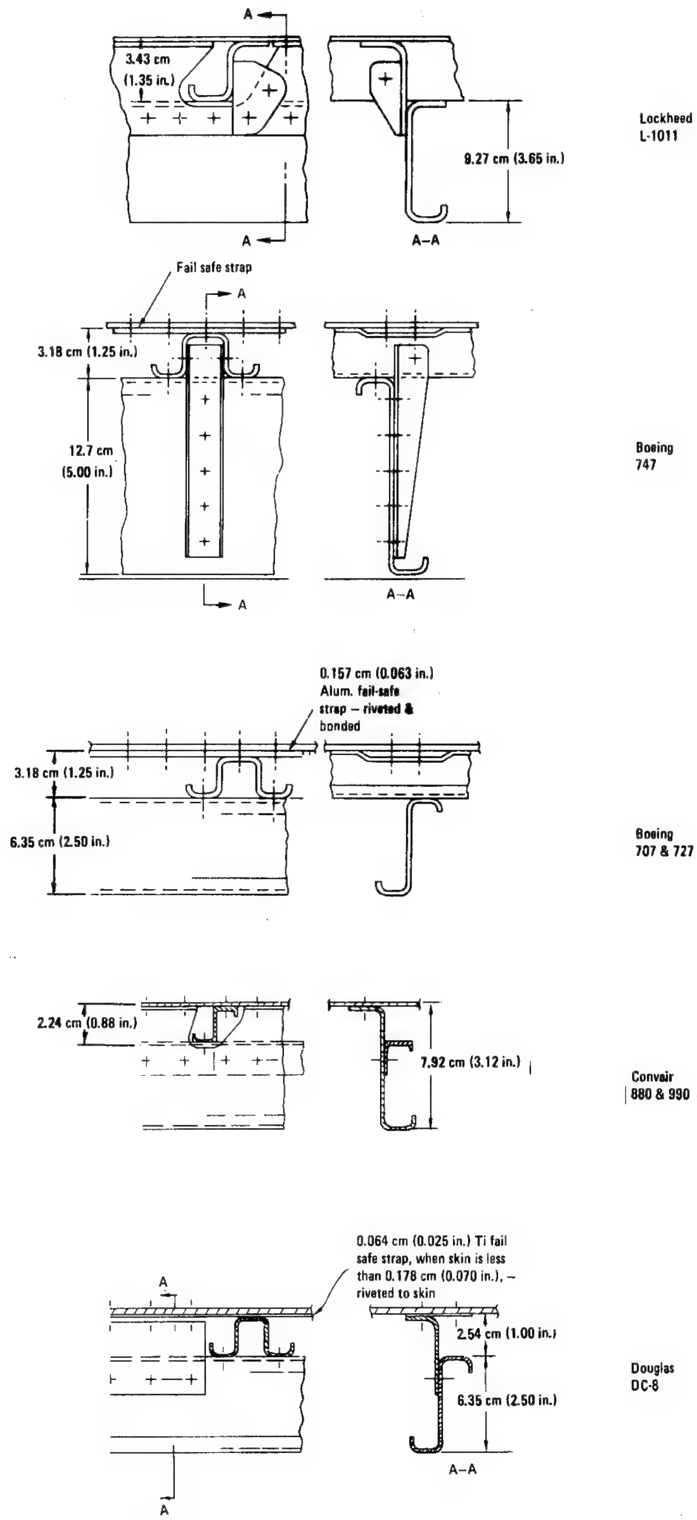


Figure 13. - Typical design for skin/stringer and frame intersection.

System Interface Requirements.- The fuselage interfaces with the environmental control system (ECS), the auxiliary power supply system (APU), the hydraulic system, the control system, the electrical system and, in some cases, the fuel and propulsion systems. The latter case would be associated with an engine located on the afterbody. For the majority of these systems, the fuselage structure must provide the necessary volume and strength to route and support the various harnesses, plumbing, and ducts associated with these systems. Provision must be made for both the normal and damaged environmental requirements of each system; for example, a ruptured bleed air line could release air at approximately 589 K (600°F) for a short duration.

Structural Interface Requirements.- Major structural components such as the nose landing gear support structure, the wing carry-through structure, the pressure deck, the cargo containment structure, and the flight station/ fuselage interface structure may pose structural requirements that could greatly impact the design of a composite fuselage. These components, in general, impose high concentrated forces on the fuselage structure, which requires reinforcement members (e.g., bulkheads, doublers, etc.) and thicker skins to redistribute these loads into the shell.

Minimum Skin Thickness.- Selection of the minimum skin thickness for the fuselage and its corresponding ply layup are influenced by several design considerations, namely, its tolerance to impact damage during manufacture and operation, its fail-safe capability combined with the capability of its adjacent structure to maintain flight safety in the event of structural damage, and its resistance to fatigue damage during the operational life of the airplane. The most critical consideration from among those listed will establish the minimum skin thickness for the fuselage shell.

Loads

The structural design loads criteria for commercial fuselages are well established and can be classified into five basic categories:

- Pressurization
- Inflight maneuver and gust
- Landing, taxi, towing, etc.
- Handling
- Emergency landing

The design loads evolving from these basic conditions are not expected to impact the composite structure to any greater extent than they do the conventional aluminum structure. For reference, the maximum limit loads for the fuselage shell of the L-1011 aircraft are shown in Figure 14. These loads represent the design axial load and shear occurring on the upper crown, sidewall and lower crown regions of the shell. The membrane forces from cabin pressurization and aerodynamic pressure are not included in these loads.

There are several load considerations that could impact the design of a composite fuselage. The inherently greater stiffness associated with composite structure could alter the wing aeroelastic loads distribution and affect the tail balancing load and hence the body bending loads on the afterbody. In addition, the lower fuselage structure must be capable of absorbing the energy associated with an emergency crash landing. The design of these areas must be carefully monitored during the design process to ensure passenger safety is maintained.

Material Properties

To provide an adequate design data base, environmental effects on the design properties of composite material systems must be assessed. The basic strength of the composite material revolves around the fiber and the matrix. It is generally recognized that the matrices of the current available composite systems require improved properties relative to their durability and ductility. In present material systems, it is the matrix which dictates the strength and durability of the part. The fiber strength capability is considerably above that of the matrix.

Material applications should be continually reviewed relative to their durability. The manufacturing capability will also impact the matrix and fiber orientation. Many new matrix combinations may emerge but the present systems include: graphite/epoxy T300/5208, graphite/epoxy AS/3501-6, graphite/epoxy T300/BP907, Kevlar/epoxy and S glass/epoxy.

The basic composite material system must be evaluated to define its: fire resistance, impact damage tolerance, repairability, ease of manufacturing shapes and assemblies, moisture and temperature capabilities, strength after exposure, and compatibility with metallics.

Experimental evidence should be provided to demonstrate that the material allowables are attained with a high degree of confidence in the most critical environment exposures, including moisture and temperature, to be expected in service. The effect of moisture absorption on static strength, fatigue and stiffness properties, for the operational temperature range, should be determined for the material system through tests. However, existing test data may be used where it can be shown directly applicable to the material system. Where existing data demonstrate that no significant temperature and moisture effects exist for the material system and construction details, within the

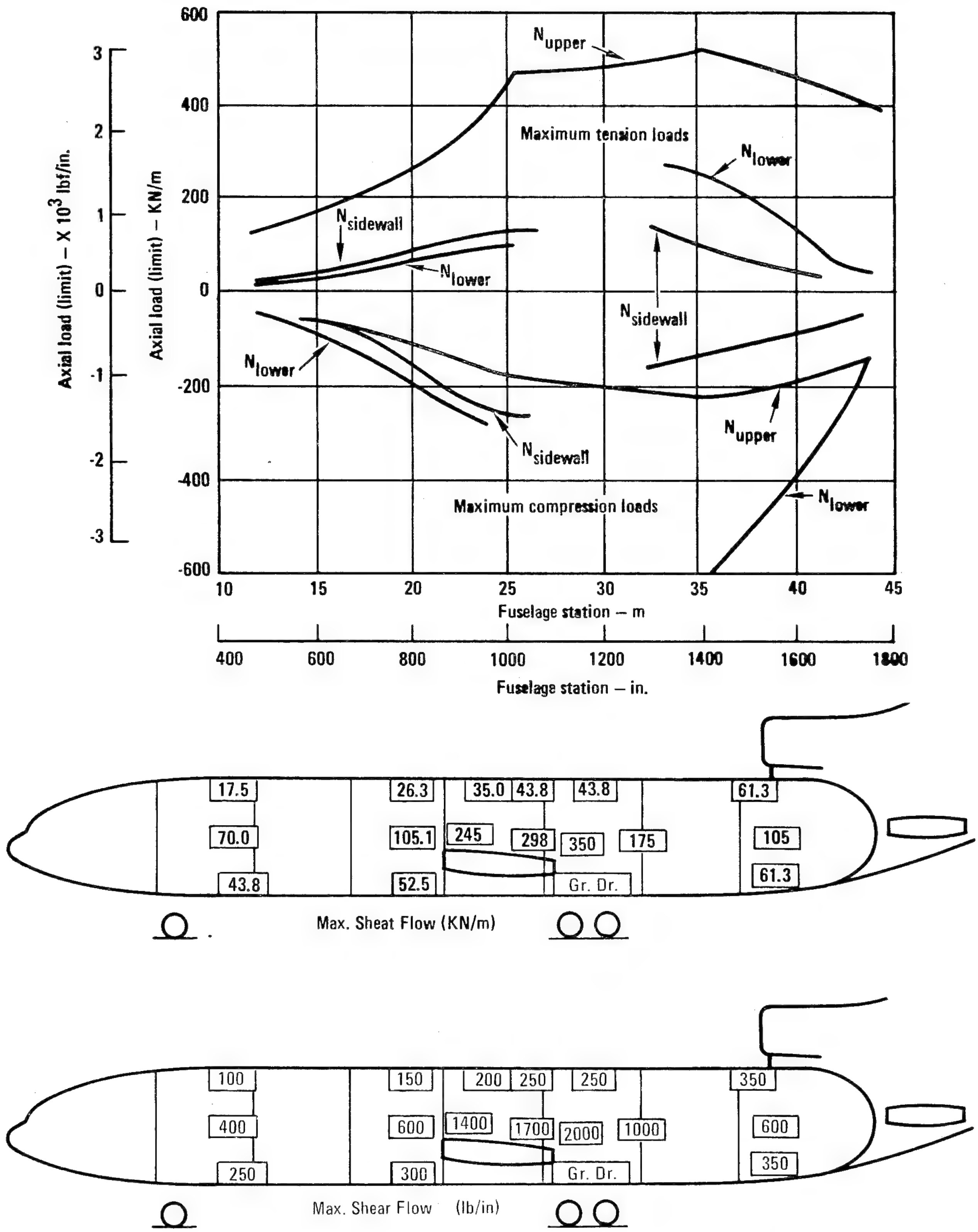


Figure 14. - Maximum limit load intensities in fuselage shell.

bounds of moisture and temperature being considered, moisture and temperature studies need not be considered.

Design Strain Levels

Design strain levels of graphite/epoxy structures are currently restricted by many considerations including: stress concentrations associated with cutouts, joints and splices; tolerance for impact damage; transverse cracking in the 90 degree fiber-oriented plies; and compatibility with adjacent aluminum strain levels. Currently, these considerations restrict the design ultimate strains to approximately fifty percent of the composite material failure strain, or about 4500 to 5000 μcm/cm, and practical working strain levels (limit load levels) from 3000 to 3500 μcm/cm.

Ultimate strain limitation.- Considerations for restricting the design strain level and some typical strain values are summarized in Table 6 and are described as follows:

Numerous studies have shown that, depending on the laminate layup, the strength of composite structures is reduced by 40 to 60 percent due to open, unreinforced, and as-fabricated holes. The reduction in strength resulting from impact damage (even below the visible detectable level) can be of the same order. The effect of these considerations reduce both the tensile and compressive strengths of a composite laminate with the compressive properties more adversely affected than the tensile properties. Since holes or damage of these types are likely during the lifetime of a structure, the design ultimate strain is usually restricted to 50 percent of that for an unflawed laminate. Typically, material failure strains are on the order of 9,000 μcm/cm; thus, considering flaws reduces the design ultimate stains by approximately 50 percent to a representative value of 4500 μcm/cm.

Most laminates contain 90° plies for the purpose of off-axis load introduction, reduction of Poisson's ratio or for biaxial loadings. Depending on curing stresses, moisture content, layup, etc., the strain level for cracking of these 90° plies varies considerably. However, about 3000 μcm/cm is the order of magnitude for onset of first ply cracking in well designed and cured laminates. Thus, a laminate with the design ultimate strain on the order of 5000 μcm/cm and a limit load strain of 3350 μcm/cm is not likely to have significant single cycle first ply cracking at limit load. Evidence to date suggest that microcracking under cyclic loading occurs at lower strain values but these also are probably held within reasonable limits. A 4500 μcm/cm design ultimate strain results in a 3000 μcm/cm limit strain and is a frequently used design assumption. Microcracking of the resin has been assumed to be the cause of increases in moisture pickup of laminates.

Under fatigue loadings with tension-compression cycles in the stress ratio (R) range from -0.5 to -1.0, fatigue failures occur in the range of

TABLE 6. - RATIONALE FOR RESTRICTING THE DESIGN ULTIMATE STRAIN LEVELS FOR GRAPHITE/EPOXY STRUCTURE

Consideration	Condition	Results	Ultimate Strain Level μcm/cm
Open Hole (Unreinforced)	Static Tension and Compression Loading	40 – 60% Strength Reduction	(0.5 x 9000) 4500
90^0 Ply Off-Axis Loading	Onset of First Ply Cracking	3000 μcm/cm	(1.5 x 3000) 4500
Fatigue Testing (Quasi-isotropric Laminates)	Tension-Compression Cycles R = -0.5 to -1.0	10^5 to 10^6 Cycles at 3300 μcm/cm	(1.5 x 3300) 5000
Non-detectable Impact Damage	Hail, Tool Drops, etc.	30 – 60% Strength Reduction	(0.55 x 9000) 5000

10^5 to 10^6 cycles in quasi-isotropic laminates at about 1/3 of ultimate strain. Similar observations were made for laminates with 67 percent of 0° fibers. The strains to failure of these two laminates were about equal to 10,000 μcm/cm even though the strengths were different. (viz about 475 MN/m^2 (69 ksi) for the quasi-isotropic and 979 MN/m^2 (142 ksi) for the 67 percent of 0° fiber laminate). This translates to a restriction of fatigue strains to about 3,300 μcm/cm. This observation does not include moisture/temperature effects.

Metallic reinforcement members may be required in the design of a composite fuselage at areas where high concentrated loads occur. At these areas, the strain level of the composite structure will have to be restricted to approximately 4500 μcm/cm to ensure that the fatigue cutoff stress of the adjacent aluminum structure is not exceeded.

Operational strain limitations.- Restrictions on the skin circumferential strain may be required because of the constant amplitude pressure load cycle which occurs on the skin during each flight in the life of the aircraft.

For aluminum fuselage skins, stress limitations of 100 MPa (14,500 psi) have been placed on the hoop stress due to fuselage pressurization (pr/t). Since very little fatigue test data are available on biaxially loaded graphite/epoxy laminates, an operational strain allowable of approximately 2200 μcm/cm appears to be a reasonable value for the composite skin in the circumferential direction.

Buckling Limitations

In the design of commercial aircraft, restrictions are placed on the postbuckling behavior of the fuselage shell to ensure adequate fatigue life during operation. These restrictions are generally applied to the initial buckling strength of the skin between stringers or longerons.

Current wide-bodied aircraft of the L-1011 type generally require that the pressurized structure be unbuckled under 1 g level flight loads in combination with normal pressure loads. In addition to this requirement, the L-1011 fuselage skins are designed such that the ultimate design shear flows do not exceed five times the initial shear buckling value, i.e., $q_{ult}/q_{cr} \leq 5$. In actual design, however, shear flows will rarely exceed three times the critical value.

Recent post-buckling fatigue tests of flat, cocured, J-stiffened composite panels under cyclic shear loading (Reference 6) indicate that panels designed to the above criteria can sustain 10^4 to 10^5 cycles at limit load (in this case, a $q/q_{CR} = 3.3$) before experiencing fatigue failures due to skin-stiffener debonding. Consequently, these requirements appear to be realistic constraints for the design of composite fuselage structure.

The post-buckling behavior of the skin in compression will generally be controlled by instability of the stiffeners or by maximum strain limitations and no additional restrictions need to be imposed on the design.

Damage Tolerance Requirements

The design of a composite fuselage must provide the necessary strength and rigidity for the structure to sustain the loads and environment imposed during operation and yet have effectively unlimited life. In meeting this goal, foremost in the aircraft designer's mind must be the provision for passenger safety. Passenger safety is maintained by formulating realistic impact criteria that define the possible types of damage that can be inflicted on the aircraft and designing a durable structure which is capable of withstanding these damages without lowering its structural integrity below a safe level.

Impact damage.- In formulating impact criteria for airplanes constructed primarily of composites, consideration must be given to a number of potential hazards that can greatly affect the integrity of the airframe. These requirements have not been stipulated on aluminum structures because of the material's inherent characteristics in withstanding most hazards satisfactorily. For example, panels made from composites, using present construction techniques, exhibit a large reduction in strength, compared with metal panels when penetrated as in the case of impact by objects.

The operational hazards, from the structural damage standpoint, include birds, hail, debris such as stones and bolts, dropped tools, engine fragments, and tire shrapnel from tread separation and tire rupture. Most of these hazards normally produce only cosmetic effects on aluminum airframes structure, with an insignificant effect on strength. On the other hand, because present composite structure is sensitive to impact damage, these hazards must be considered and rational criteria established.

Impact of composite structures can result in no damage, non-visual damage, damage not readily visible, or visible damage. Significant visible damage undoubtedly would be repaired prior to the next flight which means that the structure should be of adequate strength so that the flight can be safely completed after sustaining such damage. Damage not readily visible most likely will not be detected until the next inspection period. Under this circumstance either a safe life concept should apply or the structure should be shown to be capable of withstanding operational loads with the damage present for the number of flight hours between inspections. The effect of damage on subsequent moisture content must be accounted for because moisture, in freezing, will expand and, if entrapped in damaged composite structure, can cause additional damage.

A preliminary assessment of composite fuselage impact criteria is summarized in Table 7. The following discussion provides rationale for the various impact criteria.

Birds: To ensure that the flight crew is afforded the same level of protection from bird strike provided by the windshield, certain areas of the cockpit structure must be capable of withstanding impact of 1.8 kg (4.0 lbm) bird as stipulated in the table.

Hail: Hail encounters are characterized by multi-impacts and various size hailstones and the number of impacts is influenced by geographical location. A summary of the frequency of encounter and the impact density of hailstones is presented in Figure 15. The conditional probability for various size hailstones was obtained from Reference 5. The probability of encountering a hailstorm on the ground was obtained from the amount of time L-1011's are on the ground at various airports and the annual hail frequency for those areas. The average length of time for hail over the continental United States along with airplane fleet flight time was used for determining the probability for encountering hail during flight. Both on the ground and inflight probabilities take into account the period in the day that hail occurs and the size of the L-1011 fleet.

TABLE 7. - COMPOSITE FUSELAGE IMPACT CRITERIA

HAZARD	LOAD CONDITION AT IMPACT	VULNERABLE AREAS	CRITERIA: CUMULATIVE FREQUENCY PER 10 FLIGHT HOURS	CRITERIA: WEIGHT AND/OR SIZE	CRITERIA: IMPACT VELOCITY	CRITERIA: IMPACT ANGLE	CRITERIA: SURVIVABILITY (3)
Birds	Maximum Sea Level Cruise 25.5 kPa (3.7 psi) Cabin Pressure Differential	FS158 to 235; WL260 and Above	Not Applicable	1.8 kg (4 lbm)	179.8 m/s (590 FPS)	0.773 rad (42 Deg) to Surface	No Penetration
		FS123 to 235; WL170 and Lower				0.297 rad (17 Deg) to Surface	
		FS123 to 235; WL170 to 230				0.436 rad (25 Deg) to Surface	
Hail	On Ground	Upper Surfaces	(1)		Vel. = 86.28 $\sqrt{\text{Dia.}}$ (4)	1.57 rad (90 Deg) to Surface	
	Inflight Gusts 58.6 kPa (8.5 psi) Cabin Pressure Differential	FS158 to 235; WL260 and Above	(1)	(1) (2)	271.3 m/s (890 FPS)	0.820 rad (47 Deg) to Surface	No Penetration
		FS123 to 235; WL170 and Lower				0.419 rad (24 Deg) to Surface	
		FS123 to 235; WL170 to 230				0.593 rad (34 Deg) to Surface	
		FS235 to 449; WL280 and Above				0.297 rad (17 Deg) to Surface	
		FS235 to 449; WL135 and Lower				0.314 rad (18 Deg) to Surface	
		FS235 to 449; WL135 to 230				0.419 rad (24 Deg) to Surface	
		FS449 to 1600; WL101.5 to 219				0.209 rad (12 Deg) to Surface	
		FS449 to 1600; WL219 to 280				0.157 rad (9 Deg) to Surface	
		FS123 Bulkhead	Not Applicable			1.57 rad (90 Deg) to Surface	No Penetration
Takeoff And Landing Area Debris — Gravel, Bolts and Ice	On Ground	FS573 to 705; WL135 and Lower	Not Applicable	1.27 cm (0.50 in.) Dia. Al. Spheres. One For Each 0.092 m^2 (1.00 ft^2) of Exposed Area	60.9 m/s (200 FPS)	Varies From 0.593 rad (34 Deg) at FS14.6 m (573 in.) to 0.314 rad (18 Degs) at FS17.9 m (705 in.)	
Tools	On Ground	External Upper Surface Internal Lower Surface	Not Applicable	0.453 kg (1 lb) Hemisphere 2.54 cm (1 in.) Dia.	6.1 m/s (20 FPS)	1.57 rad (90 Deg) to Surface	•
Engine Fragments — 1/2 Fan Blade	1 ±1/2 G Flight 58.6 kPa (8.5 psi) Cabin Pressure Differential	Fuselage Sides FS1030 to FS1070 Fuselage Sides FS1045 to FS1105	Not Applicable	Maximum 2.72 kg (6 lbm) Titanium 25.4 cm (10 in.) long	365.8 m/s (1200 FPS) 410 Rad. Per Sec.	1.57 rad (90 Deg) to Surface	
Engine Fragments — 1/2 Disc		Fuselage Sides FS1085 to FS1170		Maximum 59.0 kg (130 lbm) Steel 86.4 cm (34.0 in.) 20.3 cm (8.0 in.)	126.4 m/s (415 FPS) 1110 Rad. Per Sec.		
Nose Gear Tire Shrapnel	Takeoff And Landing	Wheel Well	800	55.9 (22.0 in.) X 21.59 (8.50 in.) X 1.27 cm (0.50 in.) Rubber 2.45 kg (5.4 lbm)	64.0 m/s (210 FPS) 200 Rad. Per Sec.	1.57 rad (90 Deg) to Surface	
		Bottom FS449 to 600				1.57 rad (90 Deg) at FS11.4 m (449 in.) to 1.57 rad (20 Deg) at FS17.1 m (674 in.)	

(1) See Figure 15
(2) Density of hail 1/2 Ounce Per Cubic Inch
(3) The effect of damage on subsequent moisture content must be accounted for in structural analysis
(4) Diameter in inches

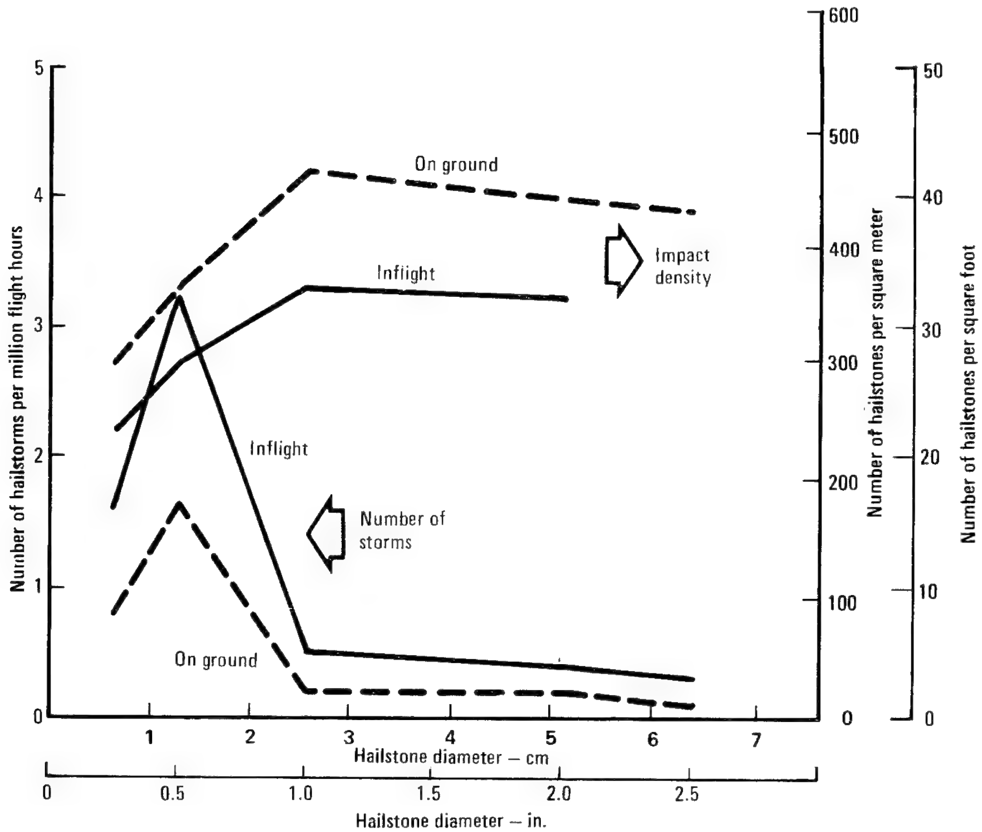


Figure 15. - Frequency of encounter and impact density of hailstones for aircraft.

The impact density of hail on the ground was obtained from Reference 7. In the absence of analogous data for the inflight case, the ground values are modified to obtain an inflight equivalent taking into account hail fall rate, the average size of the hailstorm and airplane flight speed.

To provide the necessary safety for the flight crew, hail impact in the cockpit area shall not result in penetration of the area by the hail. Inasmuch as flight hail is usually encountered in turbulent air, the airplane flight condition includes gusts as well as the appropriate cabin differential pressure.

Takeoff and landing area debris: The debris affected structural areas of the fuselage were derived from information contained in Reference 8. The size and weight of the impactor is based on a standard representation of gravel.

Tools: The tool weight is that given in Reference 9 and is dropped from a representative working height which results in the impact velocity given.

Engine fragments: Cruise power is used to determine the velocities imparted to the engine fragments. The sizes of the fragments were determined from experience and analysis. It is expected that these fragments will completely penetrate the fuselage. The hole size should not be less than 2.6 times greater than the area of the fragment.

Nose gear tire shrapnel: The size and weight of tire shrapnel is mostly dictated by thrown thread. Of the tire failures that lead to shrapnel, 80 percent are thrown treads and the remainder are carcass ruptures. The rate given in Table 7 for these type failures is based on airline experience.

It is expected that in some cases the crew will be unaware of impact from tire shrapnel during takeoff, which indicates that structure should be able to withstand operational loads until the damage (if any) is detected.

Fail-safety provisions.- A fail-safe policy must be established to ensure that flight safety is maintained in the event of structural damage of reasonable magnitude. Such damage may arise from unreported accidental impact, minor collision, turbine disc penetration, or other sources as well as fatigue.

The composite fuselage structure will be designed so that, for any specified type and level of fail-safe damage, it will sustain the prescribed loads for the conditions specified in FAR 25.

The damage-tolerance requirements for graphite/epoxy structure will be based on service experience with metal structure and a qualitative understanding of the damage-tolerance characteristics of graphite/epoxy composites. There are, however, more variables inherent in the design of composite structures which influence the types and extents of damage that are likely to occur and the modes and directions of damage propagation. Thus, it is incumbent upon the designer to anticipate the potential failure modes and failure sequences in composite structure on a part-by-part basis and to interpret the damage-tolerance requirements.

Damage-tolerance requirements will be based on the following considerations:

- The specified type of damage should have a high probability of being detected by the normal inspection procedures and intervals before the damage reaches the critical fail-safe damage size.

- The assumed damages that are likely to occur sometime during the life of the structure due to hail impact, dropped tool, minor collision, gravel impact, fatigue damage initiation and growth, etc.

For composite fuselage structure, the following types of damage must be considered in establishing a realistic fail-safe policy:

- Any single member in the substructure completely severed. For fail-safe purposes, a single member is any redundant structural member or part of any member where the remaining part can be shown to have a high probability of remaining intact in the event of the assumed failure. It must be demonstrated that the damage to the assumed severed member must be readily discoverable by normal inspection methods.

- A delamination between any two separately cured composite members which are adhesively bonded together. The extent of delamination and the effectiveness of delamination barriers must be defined.

- Delamination between individual plies at the midplane of skin surfaces and shear webs or between the skin and core of sandwich construction. The extent of delamination and the effectiveness of the delamination barriers have to be specified.

- At any location on the external surfaces, a reasonably long cut through the surface and any members integral with or bonded to the surface.

- At a cutout, a cut through the skin or web extented from the edge of the cutout to an effective damage barrier.

- All fail-safe mechanical joints and skin splices shall be designed to have sufficient shear lag capability to distribute loads from the failed section.

- For local areas of structure not meeting any of the above damage criteria, it must be shown by tests that the maximum extent of damage that is likely to be missed by a specified in-service inspection technique must not grow to a critical size for the fail-safe loading condition within prescribed inspection periods.

For the defined damage cases, it must be demonstrated by analysis and/or test that detectable damage will propagate slowly under normal operational loads so that detection and repair are ensured before reaching a critical damage size.

Fail-safe (damage tolerance) methods of analysis applicable to composite structure are generally not available. Therefore, compliance with the fail-safe policy will most likely be based on fail-safe testing of sub-components and full-scale structure. In some cases finite element analyses must be used to determine the redistribution of loads around damaged regions.

Fatigue Consideration.- The basic fatigue policy for a composite fuselage is that the structure shall not be life limited in operational service. This means that with normal operation, inspection, maintenance and repair, it is intended that the ultimate retirement of the structure, when it occurs, would be for reasons other than structural fatigue or corrosion. An unlimited life structure can be achieved by proper choice of materials and processes, design stress (or strain) levels, detail design quality and adequate protection against lightning and foreign object damage.

The probable issues involved in the fatigue evaluation of a composite fuselage will include an assessment of the strain levels which causes matrix cracking and/or delamination. This assessment will most likely include the establishment of strain cut-off levels for the hoop pressure strain for the operating condition and for the ultimate design condition. Additional strain criteria must be formulated for design regions that are not inspectable and areas with high stress considerations.

Similar to metallic structure, verification of the life of the composite structure must be demonstrated by fatigue analysis and fatigue testing. This testing could involve the fatigue testing of elements, components, and large-scale articles to a loading spectrum simulating the operational environment.

Acoustic Considerations

This section discusses the general acoustic environment on a composite fuselage and its influence on cabin noise level and the structural design of the airframe. Experience with current high performance aircraft has demonstrated the necessity of a coordinated program of development testing and analysis to assess the impact of the acoustic environment on the design of the structural configurations. Starting this program early in the design stage will allow sufficient time to make adjustments in the design of the composite structure so that the required noise transmission characteristics and sonic fatigue properties can be attained.

Acoustic transmission.- Current jet transport aircraft require only modest acoustic treatment with a minimal weight penalty to achieve comfortable cabin interior noise levels. Average interior noise levels will typically range from 75 to 80 dBA, with worst seat values (usually a window seat) running 2 to 5 dBA higher. Maximum overall sound pressure level (OASPL) values tend to range in the neighborhood of 95 dB and speech interference level (SIL) values of 65 to 70 dB.

For turbofan aircraft at high-speed cruise, M = 0.85 at 9144 m (30,000 ft) altitude, the interior noise levels are governed by transmitted turbulent boundary layer noise. The typical exterior noise levels are 135 dB (OASPL) with peak one-third-octave band sound pressure level (SPL) values of 125 dB. Boundary layer noise reduction values of the order of 30 to 50 dB are needed for the one-third-octave bands with center frequencies below 1000 Hz.

If a strength designed composite fuselage results in a large reduction in structural wall mass relative to current metal fuselages, then a substantial increase in the surface density of the acoustic treatment may be required, expecially if the interior noise goal of 80 dBA is to be met (Reference 10). The need for increased density treatment becomes more acute towards the aft end of the fuselage where the external boundary layer is thicker.

Composite fuselage structure should be reviewed for their noise transmission characteristics. Factors that can influence these characteristics include stiffener spacing, skin thickness and stiffener section properties. In addition, noise control elements such as trim panel mass, spacing between double walls, outerwall mass, viscoelastic outer wall damping treatment and fiberglass blanket insulation can reduce the interior noise level. Advanced noise reduction design methods should be explored to assess their effect on interior noise. Such design methods include increasing outer wall stiffness with minimal outer wall mass increase, varying the trim panel mass in conjunction with the first method, reducing the working stress by increasing the outer wall stiffness, and studying the viscoelastic damping treatment in conjunction with the above.

Sonic fatigue.- Generally, the pressurized fuselage structure on current widebodied transport aircraft which are powered by large turbofan engines is designed by considerations other than sonic fatigue. The maximum jet noise levels on the fuselage are generally low, with the highest levels occurring on the rear fuselage at takeoff (Figure 16). The highest noise levels on the forward fuselage occur during landing when the engine thrust is reversed, just ahead of the wing root leading edge. Typical design life requirements at maximum takeoff and reverse thrust levels are 360 and 150 hours, respectively, for the life of the aircraft. These design life requirements represent a range of cycles from 5×10^7 to 3×10^8 depending on the structural resonant frequencies.

The amount of curvature of the shell could greatly affect the sonic fatigue life of the structure, i.e., the greater the curvature the lower the stresses; hence, an increase in life. For preliminary design purposes, a sonic fatigue analysis of a simple flat panel representative of the fuselage structure is most likely sufficient for establishing the lower bound for the sonic fatigue life.

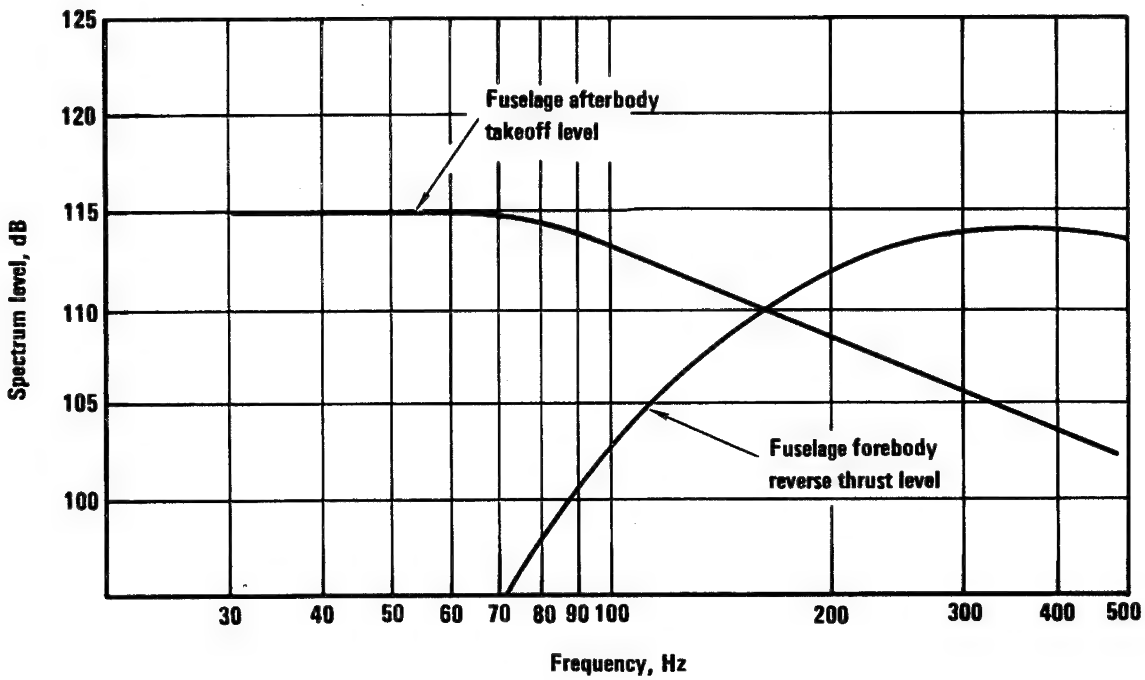


Figure 16. - Typical fuselage design environment.

The sonic fatigue allowables (root mean square strain level for a required life) for graphite/epoxy are considerably greater than those for the aluminum alloys currently used. The actual sonic fatigue allowables used depend on the details of the design such as the attachment method (bonded or riveted), and the ply orientation of the laminate used. The existing random fatigue data for composites are summarized in Figures 17 and 18 for fastener-attached and bonded joints as well as for the basic laminate ($K_t = 1$). The random fatigue curves become flat after approximately 5×10^7 cycles, which is in the design range for the fuselage.

Crashworthiness

The design of a composite fuselage must assure that occupants have every reasonable chance of escaping serious injury under realistic and survivable crash conditions. The use of composite structures in areas where failure would create a hazard to occupants should be shown to have crashworthiness capability equivalent to conventional structural materials. In general, this equivalency would be shown by comparative analysis supported by tests as required.

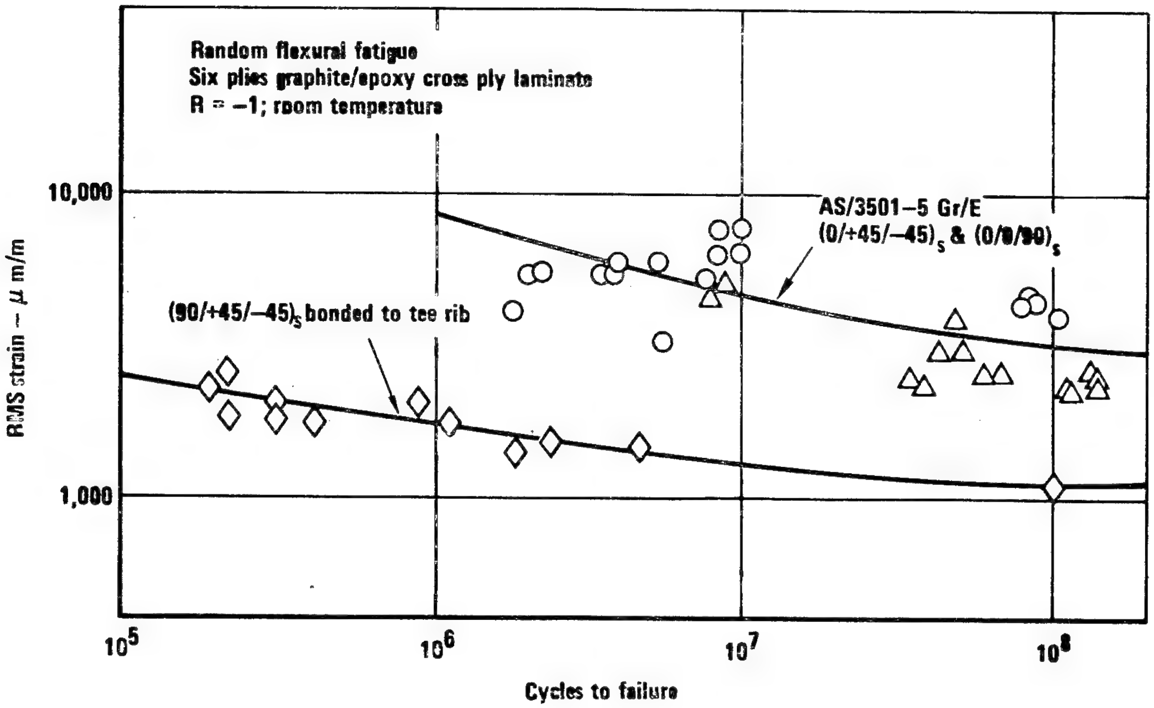


Figure 17. - Test coupon random fatigue data.

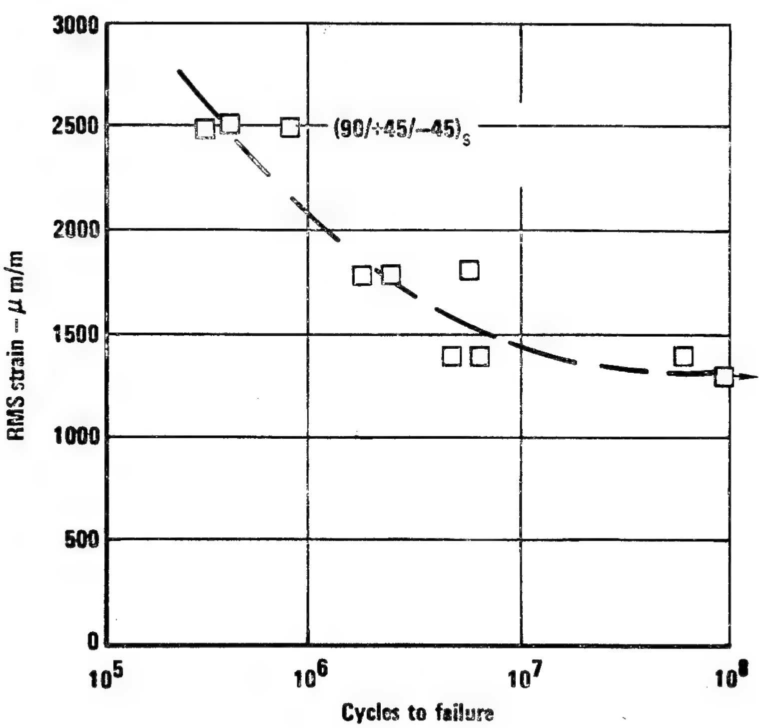


Figure 18. - Coupon random fatigue data - composite skin attached to composite stiffener with fasteners.

The design of a composite fuselage shell at inception to meet the equivalent-to-aluminum goal requires the development of an extensive design data base. Investigations of structural response and integrity of composite fuselage structural components subjected to selected crash events are required. In particular, the design of the fuselage lower crown structure for crashworthiness considerations may significantly influence the design of the complete pressure shell, including damage tolerance requirements.

MANUFACTURING CONSIDERATIONS

The design of cost-competitive hardware requires the integration of key manufacturing considerations in the design process. The large components, complex tooling and equipment requirements associated with the manufacture of a producible fuselage structure will have significant cost impacts. Manufacturing considerations must also include quality assurance considerations to ensure the integrity of the fabricated hardware. Manufacturing of composite components are configuration sensitive and must be performed in conjunction with the structural design effort.

Materials/Material Cost

The design of large fuselage components employing graphite/epoxy composites will require the use of materials in various forms. For skin and stiffener designs the use of preplied and unidirectional tape will find application. For frame components woven cloth can be employed, with unidirectional tape used for flange reinforcement.

Fabrication Costs

The design of fuselage shell components that are low cost will require the use of net resin system and cocuring the major assemblies. Although concepts for a unitized shell and frame assembly have been explored, for practical considerations separate skin-stringer and frame assemblies will most likely be the near term fabrication approach for the design of a composite fuselage. The attachment method for joining these assemblies will have a major cost impact. Bolt-bonding application could result in a reduction of the number of fasteners and improved fatigue life.

Tooling Requirements

The design of the structural elements will reflect on the tooling concepts employed. The tooling must facilitate locating and supporting the prepreg stiffener during the cure cycle. Tooling for cocured frame assemblies should be designed to facilitate cocured net molded parts to eliminate

contour machining requirements. Elastomeric mandrels can be employed if the design requires the application of transverse pressure on formed parts such as the flange section of frames.

Equipment Requirements

The design of the fuselage components/assemblies should be directed to eliminate and/or minimize the major handwork labor. The utilization of automatic production machines such as: (1) roll-forming machines to form prepreg stiffeners, (2) numerically controlled (N/C) tape laying machines for skins, and (3) N/C water jet and/or Gerber cutter machine to cut out frame patterns from preplied tape and/or cloth material will reduce fabrication costs.

Producibility

The design of composite material fuselage components must follow guidelines for producing cost-competitive hardware. Specifically, (1) to develop structural shapes, sheets, and assembly configurations with corresponding fabrication approaches which are directed to facilitate minimum cost (e.g., minimum number of parts and fabrication operations); and (2) to advance the state-of-the-art in fabrication methods technology to produce hardware meeting program goals of cost, quality and reproducibility.

Some of the producibility considerations are:

- Design skin laminates to facilitate N/C automatic tape laying machines.
- Design constant cross section stiffeners (hat, zee and I sections) to facilitate automatic roll forming machines. Doublers can readily be applied to high load areas as a separate operation.
- Design for maximum use of cocuring which will eliminate fasteners.
- Design for bolt-bond joints to reduce fastener count and provide for improved fatigue life of interface joints.
- Design components to utilize preplied tape materials (e.g., combinations of from 2 to 6 plies in a given stacking sequence) for the skins, doublers, and stiffeners. Also, utilize woven cloth materials for contoured parts such as frames.

SERVICE AND ENVIRONMENTAL CONSIDERATIONS

The design of future commercial transport is prompted by the escalating fuel price and the need to apply advanced technology to keep fares and direct operating costs in line. This is also accompanied by a need to keep maintenance cost and airplane reliability at least comparable to the current advanced technology fleet. To achieve this reliability goal, airframe design criteria are established to ensure an airplane of unlimited life for primary airframe components while meeting all of the requirements for strength and stiffness, and the fail-safe or safe-life characteristics necessary for flight safety. The airframe must meet all requirements established by the Federal Aviation Administration (FAA) and the manufacturer, which encompass structural strength (both limit and ultimate) and dynamic stability (freedom from flutter). The airframe must also meet reliability requirements established by airline operators to provide for minimum maintenance time and cost, freedom from delamination and cracks, and ease of inspection and repair.

Composite structures introduce a need to address considerations heretofore not necessary for metallic airframe design as well as differing approaches to handle problems in the service environment.

The typical kinds of damage which occur on fuselage structures in service must be evaluated and considered in the design of the composite fuselage structures. A survey on airline damage experience was conducted by Lockheed-California Company for a NASA program on composite repair (Reference 11), and the following conclusions were drawn from airline responses:

- Airline damage results primarily from impact, fatigue, and corrosion. The lower fuselage is the most damage-prone area, subject to both impact and corrosion.
- The relative proportion of these causes of damage varies significantly from one airline and aircraft type to another, with the incidence of fatigue and corrosion highly dependent on aircraft service history.
- Most impact damage is from ground handling rather than inflight damage.
- The introduction of composites can be expected to eliminate corrosion and greatly reduce fatigue as causes of damage. The response to impact damage for composites will be different from metal, resulting primarily in internal delaminations which may not have associated visual indications.

The frequent incidence of ground-handling damage in the lower fuselage area indicates that damage tolerance and repairability are prime considerations in the design of a composite fuselage. The NASA program mentioned above on composite repair also included a survey of available data on composite damage tolerance. Significant conclusions derived from this survey include the following:

- Available analytical and experimental data on composite strength reductions resulting from flaws are primarily based on small coupon specimens, and primarily relate to idealized flaws such as holes and slots. Some data are available on realistic flaws such as delaminations resulting from impact. Most available data are for tensile loading conditions.

- Correlations between analytical predictions and experimental data are reasonably good in most cases.

- Composite tensile strengths are reduced as flaw size increases, with roughly 50 percent strength loss at 1.0 cm flaw sizes. As flaw size increases further, additional strength reductions occur at a more gradual rate, leveling off in the 40-60 percent range. This assumes that flaw size does not increase to a level where net section effects are introduced.

The above data are based primarily on components which are less highly loaded than fuselage components. Thus, the effect of fatigue cycling, particularly if it is compression dominated, will need to be evaluated for composite fuselage components. Also, the design ultimate strengths for fuselage components are likely to be at a level where repairs will be required to restore design strength. This significantly increases the criticality of repairs for fuselage skins compared with repairs for lightly loaded composite parts. The following points must be considered in the development of composite repair procedures. These are based on the airline survey discussed previously in which information on airline maintenance procedures were obtained:

- All repairs are considered permanent repairs and must, according to FAA regulations, restore the full design strength of the components, as well as its full fatigue life capability. This includes repairs made at line stations away from the major maintenance bases.

- The only type of repair which can be accomplished at line stations is mechanically attached external patches. This type of repair must, therefore, be developed for composites. Potential problems with line repairs are galvanic corrosion and damage caused by drilling operations.

- Airlines use both aerodynamically smooth and raised doublers for repairs, with widely varying proportions of each. The forward fuselage is an aerodynamically critical area where flush repairs are required in most cases.

- Some airlines have excellent capabilities for making structural repairs by bonding, but others completely lack facilities and experience and send removable components to outside vendors for repair.

- Airlines have virtually no experience or capabilities for on-aircraft bonded repairs, which would be required for fuselage parts.

- Maintenance down times between major overhauls are too short in some cases to accomplish bonded repairs.

- In many cases back-side access is limited because of adjacent sub-structure. Repairability is thus a critical consideration for development of a composite fuselage design. Repair procedures must be developed which are compatible with airlines capabilities, meet specialized requirements such as aerodynamic smoothness, and restore design strength and fatigue life for highly loaded primary structure.

The use of composite structures introduces a problem in inspectability. Damage often results in internal delaminations which are not visually detectable. Airlines typically use NDI procedures only to verify the extent of visual damage. It is thus possible, under current airline practices, for impact damage to remain undetected. This is not a problem for lightly loaded structures where tests have shown that damage growth does not occur. For fuselage skins, however, this could represent a problem, and new requirements may have to be imposed.

CONCLUSIONS

The objective of the study described in this report was to identify the principal design drivers associated with the design of a composite fuselage for a commercial transport aircraft. This objective was attained by (1) reviewing the many structural, manufacturing and environment and service considerations involved in the design of pressurized fuselage structure, (2) selecting the principal design drivers from these considerations and discussing their impact on the design of such a fuselage structure, and (3) examining and, wherever possible, establishing new design criteria (in principle).

A summary of the results of this study is presented in Table 1. The principal design drivers among the structural considerations reviewed during this study are: the damage tolerance aspects (impact criteria and fail-safe requirements), material properties, design features (shell cutouts, joints, etc.), post-buckling limitations and crashworthiness. The principal drivers from the manufacturing considerations are: the material cost, fabrication cost and producibility. Safety and reliability, maintainability, inspectability and repairability are deemed the principal design drivers among the service and environmental considerations.

Those considerations requiring new criteria or methodology prior to designing a composite fuselage structure include:

(1) The definition of climatogical data and interpretation of these data to define reasonable temperature and humidity profiles for design.

(2) An assessment of new and existing lightning protection methods to arrive at lightweight design which will meet CAA and FAA requirements.

(3) The establishment of hail impact criteria (size, number of impacts, areas of impact, etc.) and an assessment of the resultant damage.

(4) Requirements to quantify the effects of cutouts, joints, impact damage and transverse cracking on the design strain levels.

(5) The formulation of realistic impact criteria covering all possible types of damage that can be inflicted on the aircraft to establish damage tolerance and fail-safe policies.

RECOMMENDATIONS

The primary purpose of a fuselage of a commercial transport aircraft is to provide safe and comfortable accommodations for passengers during flight. In this respect, it is first and foremost an environmentally controlled pressurized shell which must be designed to be highly damage tolerant and crashworthy. Passenger comfort and acceptability dictate low levels of noise and vibration. Design data to address these issues are essential before the weight savings benefits of composite materials can be fully exploited.

There are a number of technical issues and potential problems areas which must be resolved before sufficient confidence is established to commit composite materials for application to pressurized fuselage structures. The key issues are identified below:

- Composite Fuselage Design Specification - Prior to the documentation of the design criteria and the structural requirements, investigations must be conducted to attain a state of design readiness. This state of readiness includes (1) a thorough understanding of the principal design drivers associated with the design of a composite fuselage, (2) a delineation of the major design problems, and (3) the development of the necessary design data base to assess and solve these problems.

- Damage Tolerance - A criterion on damage tolerance must be established to provide for passenger safety during the operational life of the aircraft. The potential operational hazards and the sensitivity of the composite structure to these hazards must be quantified in order to establish such items as the extent of damage, the associated strength level and, if required, the inspection and repair periods. These and other considerations must be defined in order to establish a rational fatigue and fail-safe policy for composite fuselage structure.

- Crashworthiness - Composite structures must be shown to have the crashworthiness capability equivalent to those of conventional aluminum structure. To attain this equivalency, a design data base must be established by conducting both analytical and experimental investigations, exploring the structural response and integrity of composite structure subjected to simulated crash events.

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16. ABSTRACT

A study was conducted to explore the structural, manufacturing, and service and environmental considerations that could impact the design of composite fuselage structure for commercial transport aircraft application. An assessment of the severity of these considerations were made and the principal design drivers delineated. A number of technical issues and potential problem areas which must be resolved before sufficient confidence is established to commit to composite materials were defined. The key issues deal with: definition of composite fuselage design specifications, damage tolerance and crashworthiness.

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